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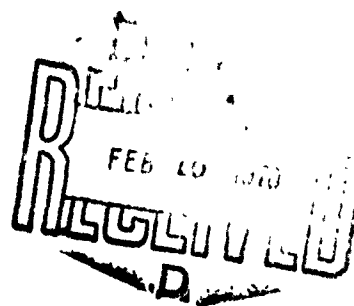
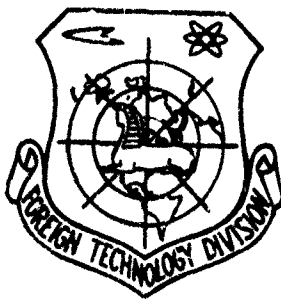
## FOREIGN TECHNOLOGY DIVISION



### PRACTICAL AERODYNAMICS OF THE An-2 AIRCRAFT

by

M. N. Shifrin



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# EDITED MACHINE TRANSLATION

PRACTICAL AERODYNAMICS OF THE An-2 AIRCRAFT

By: M. N. Shifrin

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# U. S. BOARD ON GEOGRAPHIC NAMES TRANSLITERATION SYSTEM

Block	Italic	Transliteration	Block	Italic	Transliteration
А а	<i>А а</i>	A, a	Р р	<i>Р р</i>	R, r
Б б	<i>Б б</i>	B, b	С с	<i>С с</i>	S, s
В в	<i>В в</i>	V, v	Т т	<i>Т т</i>	T, t
Г г	<i>Г г</i>	G, g	У у	<i>У у</i>	U, u
Д д	<i>Д д</i>	D, d	Ф ф	<i>Ф ф</i>	F, f
Е е	<i>Е е</i>	Ye, ye; E, e*	Х х	<i>Х х</i>	Kh, kh
Ж ж	<i>Ж ж</i>	Zh, zh	Ц ц	<i>Ц ц</i>	Ts, ts
З з	<i>З з</i>	Z, z	Ч ч	<i>Ч ч</i>	Ch, ch
И и	<i>И и</i>	I, i	Ш ш	<i>Ш ш</i>	Sh, sh
Й й	<i>Й й</i>	Y, y	Щ щ	<i>Щ щ</i>	Shch, shch
К к	<i>К к</i>	K, k	Ъ ъ	<i>Ъ ъ</i>	"
Л л	<i>Л л</i>	L, l	Ы ы	<i>Ы ы</i>	Y, y
М м	<i>М м</i>	M, m	Ь ь	<i>Ь ь</i>	'
Н н	<i>Н н</i>	N, n	Э э	<i>Э э</i>	E, e
О о	<i>О о</i>	O, o	Ю ю	<i>Ю ю</i>	Yu, yu
П п	<i>П п</i>	P, p	Я я	<i>Я я</i>	Ya, ya

\* ye initially, after vowels, and after ъ, ы; e elsewhere.  
 When written as ѣ in Russian, transliterate as ye or ѣ.  
 The use of diacritical marks is preferred, but such marks  
 may be omitted when expediency dictates.

FOLLOWING ARE THE CORRESPONDING RUSSIAN AND ENGLISH  
DESIGNATIONS OF THE TRIGONOMETRIC FUNCTIONS

Russian	English
sin	sin
cos	cos
tg	tan
ctg	cot
sec	sec
cosec	csc
sh	sinh
ch	cosh
th	tanh
cth	coth
sch	sech
csch	csch
arc sin	$\sin^{-1}$
arc cos	$\cos^{-1}$
arc tg	$\tan^{-1}$
arc ctg	$\cot^{-1}$
arc sec	$\sec^{-1}$
arc cosec	$\csc^{-1}$
arc sh	$\sinh^{-1}$
arc ch	$\cosh^{-1}$
arc th	$\tanh^{-1}$
arc cth	$\coth^{-1}$
arc sch	$\operatorname{sech}^{-1}$
arc csch	$\operatorname{csch}^{-1}$
<hr/>	
rot	curl
lg	log

Practical Aerodynamics of the An-2 Aircraft,  
M. N. Shifrin. Publishing House "Transport"  
1968, pp. 1-176.

The book discusses basic properties and laws of motion of the air, aerodynamic forces of the wing, peculiarities of layout and aerodynamic properties of the aircraft, equilibrium, stability and controllability of the aircraft under different flight conditions, and peculiarities of flight under conditions of icing.

The book is intended for students of colleges and schools of civil aviation and can be used by flying personnel of other departments operating the An-2 aircraft. There are 6 tables and 105 figures.

## INTRODUCTION

Aerodynamics is the science concerning the laws of motion of the air and its influence on a streamlined body.

An outstanding role in the creation of principles of aerodynamics and its further development belongs to Soviet scientists.

Aerodynamics is divided into theoretical, experimental and applied areas.

Theoretical aerodynamics is the science about general rules of the motion of gases and the action of these gases on solids. It has been developed on the basis of contemporary achievements in mathematics, physics and experimental aerodynamics.

Experimental aerodynamics studies the motion of the air and its force actions on bodies by means of carrying out of experiments with the help of special instruments in aerodynamic laboratories or by investigating aircraft directly in flight. Such investigations are called flight tests.

Applied aerodynamics, using data of theoretical and experimental aerodynamics, develops the theory of flight of the aircraft and creates methods of aerodynamic calculation, designing and flight tests of aircraft.



The division of applied aerodynamics which covers questions of the calculation and designing of an aircraft and its component parts is called the aerodynamics of an aircraft.

The aerodynamics of an aircraft in combination with the theory and technique of flight in reference to a specific aircraft is called practical aerodynamics of an aircraft.

Deep knowledge of the practical aerodynamics of an aircraft is the guarantee of rapid mastery of the technique of piloting and ensures flight safety.

## BASIC SYMBOLS

Wing area.....	S	Flying and gliding range,	
Wing span.....	$\ell$	takeoff distance and landing	
Length of chord of the wind.....	b	run, length of takeoff and	
Relative thickness of profile of		landing distance.....	L
the wing.....	c	Power and reaction.....	N
Relative airfoil camber,		Number of revolutions of the	
coefficient of friction.....	f	crankshaft of the engine and	
Aspect ratio of the wing.....	$\lambda$	overload.....	n
Angle of attack.....	$\alpha$	Advance ratio and force of	
Angle of flight path of the		windmilling of propeller.....	A
aircraft with the horizon.....	$\theta$	Blade angle of propeller.....	$\phi$
Angle of roll.....	$\gamma$	Efficiency of propeller (eff)...	n
Angle of deflection of control..	$\delta$	Frictional force and force of	
Angle of sideslip.....	$\beta$	inertia.....	F
Force opening the slat.....	x	Weight.....	G
Full aerodynamic force.....	R	Impact pressure and fuel con-	
Drag.....	Q	sumption per kilometer.....	q
Lift.....	Y	Consumption of fuel per hour....	$C_h$
Lateral aerodynamic force.....	Z	Specific fuel consumption.....	$C_e$
Coefficient of full aerodynamic		Tractive force.....	P
force.....	$c_R$	Radius.....	r
Drag coefficient.....	$c_x$	Acceleration.....	j
Lift coefficient.....	$c_y$	Air density.....	$\rho$
Speed.....	V	Mass.....	m
Vertical velocity.....	$V_y$	Pressure.....	p
Resultant speed.....	W	Center of gravity.....	LT (CG)
		Center of pressure.....	LD (CP)

Peripheral velocity..... u Degree of controllability,  
 Altitude of flight, propeller downwash angle.....  $\epsilon$   
 pitch..... H Mean aerodynamic chord..CAX (MAC).  
 Lift-drag ratio..... K

Note. 1. If the given symbol has several values, then to it the appropriate subscript is added. For example, the speed necessary for lift -  $V_{\text{ПОД}}$  maximum speed -  $V_{\text{МАКС}}$  aspect ratio angle -  $\theta_K$ .

2. The  $\Delta$  (delta) sign indicates an increase or difference in quantities before which it is put ( $\Delta V$ ,  $\Delta Y$ ,  $\Delta N$ ).

Greek Letters Found in the Text

$\alpha$	$\theta$
$\beta$	$\lambda$
$\gamma$	$\kappa$
$\Sigma$	$\rho$
$\Delta, \delta$	$\phi$
$\eta$	$\epsilon$

## CHAPTER I

### BASIC PROPERTIES AND LAWS OF MOTION OF THE AIR

#### 1. Basic Parameters of the Air

Air constitutes a mixture of different gases, and in a state motionless it is characterized by these parameters: pressure (p), temperature (T) and density ( $\rho$ ).

Pressure is the force acting perpendicularly on a unit of surface. Accepted for the unit of pressure is technical atmosphere (at) - pressure equal to one kilogram per square centimeter ( $\text{kgf/cm}^2$ ). According to the International System of Units SI, pressure is measured in newtons per square meter ( $\text{N/m}^2$ ) - see appendix.<sup>1</sup>

The pressure caused by a mass of higher layers of air and impacts of chaotically moving molecules of it is called atmospheric pressure and is usually measured in millimeters of the height of a column of some liquid.

The pressure in one  $\text{kgf/cm}^2$  is balanced by a column of mercury with a height of 735.6 mm:

$$p = \frac{B}{735.6},$$

where B - atmospheric pressure expressed in mm Hg.

---

<sup>1</sup>In connection with the introduction of the International System of Units SI (dated 1 January 1963, GOST (All Union State Standard) 9867-61) the appendix to training manual gives a table of units of measurements and conversion factors; the examples shows the method of conversion of units of measurement from the technical system MKGSS into the International System SI.

Temperature - degree of heat - characterizes the speed of random motion of the molecules: the higher the temperature, the more rapid the molecules move and vice versa.

The measurement of temperature can be produced on two scales: Centigrade and Kelvin. At  $0^{\circ}$  on the centigrade scale the melting point of ice is accepted and at  $100^{\circ}$  - boiling point of water at a pressure in both cases equal to 760 mm Hg.

Zero degree on the Kelvin scale is accepted as the temperature at which random (thermal) motion of the molecules is ceased. Such a temperature corresponds to  $-273^{\circ}$  on the centigrade scale. Temperature in degrees centigrade is designated  $t^{\circ}\text{C}$  and in degrees Kelvin -  $T^{\circ}\text{K}$ . The connection between them is expressed by the formula

$$T^{\circ}\text{K} = t^{\circ}\text{C} + 273.$$

The density of gas is the mass included in a unit of volume. It is determined by the formula

$$\rho = \frac{m}{v},$$

where  $m$  - mass,  $\text{kgf}\cdot\text{s}^2/\text{m}$ ;  $v$  - volume of air,  $\text{m}^3$ .

With the change in pressure or temperature the volume of the gas is changed and, consequently, its density is changed also. In connection with this, for determining the air density it is expedient to use the formula

$$\rho = 0,0473 \frac{B}{T},$$

where  $B$  - pressure, mm Hg;  $T$  - temperature of air,  $^{\circ}\text{K}$ .

## 2. Basic Physical Properties of Air

Air is characterized by such physical properties as color, transparency, heat, electroconductibility, inertness, viscosity and compressibility.

The considerable effect on forces appearing with the motion of bodies in air is exerted by inertness, viscosity and compressibility.

Inertness is the ability to resist a change in state of the relative rest or rectilinear uniform motion. A measure of inertness is density, with an increase in which the inertness of air is increased.

Viscosity is the ability of air to resist a mutual shift in its own particles. The cause of viscosity is forces of internal friction of particles against each other with their mutual shift.

Compressibility is the ability of air to change its volume and, consequently, density with a change in pressure and temperature.

Compressibility is characterized by the ratio of change in density ( $\Delta\rho$ ) to change in pressure ( $\Delta p$ ), i.e., quantity  $\Delta\rho/\Delta p$ . The greater the ratio, the greater the compressibility and conversely.

Thus, a change in parameters of the air by means of a change in physical properties of it has an influence on the magnitude of those forces which appear with the motion of a body in air.

Therefore, the same aircraft tested in different places of the earth (with different parameters of the air) or tested at a different time of the day under other identical conditions will indicate different flying data.

For a comparison of aircraft with respect to their flying characteristics, results of the tests lead to standard conditions for which the concept of the International Standard Atmosphere (ISA) is introduced.

The International Standard Atmosphere is accepted by all countries of the world as a table of changes in basic parameters of the air with a change in flight altitude.

The data in the ISA table approximately correspond to the average annual parameters of the air in the middle latitudes of the earth (Table 1).

Table 1.

Altitude m	Pressure mm Hg	Temperature, °C	Density, kgf·s <sup>2</sup> /m <sup>4</sup>	Altitude m	Pressure mm Hg	Temperature, °C	Density, kgf·s <sup>2</sup> /m <sup>4</sup>
0	760	15.0	0.125	2500	560	-1.3	0.096
500	716	11.8	0.119	3000	526	-4.5	0.093
1000	674	8.5	0.113	3500	493	-7.8	0.088
1500	634	5.3	0.108	4000	462	-11.0	0.084
2000	596	2.0	0.103	4500	433	-14.3	0.079

Note. Subsequently, flight altitude will mean the altitude corresponding to air density according to the standard atmosphere.

### 3. Airflow and Forms of Its Motion

Airflow is called in a definite way a directed flow of mass of air. According to the character of motion, airflow is subdivided into two forms, steady and transient.

The motion at which parameters of air at each point of flow remain constant in the course of time, although unequal at various points of it, is called steady.

The motion at which parameters of airflow at a given point in the course of time are changed is called unsteady. The overwhelming majority of motions of air encountered in nature is unsteady.

The study of unsteady motion is considerably complex, in view of which in aerodynamics the steady motion of air is examined, which does not reduce the practical importance of conclusions made in the study of basic laws of aerodynamics.

#### 4. Law of Constancy of Flow Rate of Air Per Second

The law of constancy of the flow rate of air per second is the most important law of aerodynamics; it is based on the law of the conservation of matter, the essence of which consists of the fact that matter is eternal, and it is not created and does not vanish tracelessly.

The application of this law of nature to the stream of an ideal gas is the law of constancy of flow rate per second, which is formulated in the following way.

With the steady motion of gas through any cross section of a given stream in one second the same mass of gas passes.

An ideal gas is called a gas not possessing the property of viscosity.

Figure 1 shows a stream of variable cross section. The volume of air passing through sections of stream I-I and II-II in 1 second can be determined by the product of the area of the section ( $F$ ) by the velocity of the stream in the given section ( $V$ ). The product of a volume of air per second by its density ( $\rho$ ) determines the per second mass flow rate of air.

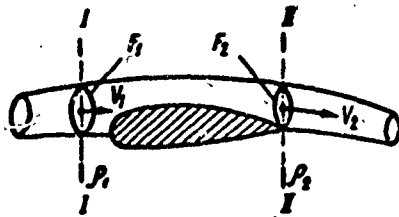


Fig. 1. Steady motion of a stream of ideal air.

According to law of conversation of matter, the mass of air  $m_1$  passing through section  $F_1$  and the mass of air  $m_2$  passing through section  $F_2$  in 1 second, are equal to each other, and the magnitude is constant:

$$m_1 = m_2 = \text{const} \quad \text{or} \quad \rho_1 V_1 F_1 = \rho_2 V_2 F_2 = \text{const.}$$



Under the condition that the air is incompressible and its density is constant ( $\rho_1 = \rho_2 = \text{const}$ ), this permits simplifying the equation and writing it in the following way:

$$V_1 F_1 = V_2 F_2 = \text{const.}$$

From the given equation this very important practical conclusion follows: the smaller the cross section of the given stream, the greater the airspeed in it, and, conversely, the larger the cross section of the stream, the lower the speed in it.

Thus, the law of constancy of the flow rate of air per second sets the dependence between the section of the stream and airspeed, which comprises its physical essence.

## 5. Bernoulli Law

The Bernoulli law is the application of the law of conservation of energy to the stream of an ideal gas, the essence of which consists of the fact that energy does not vanish tracelessly and does not appear from nothing but can only pass from one form to the other.

Figure 2 shows the stream of an ideal air under the condition when there is no exchange of energies between the stream and environment. On the basis of such an assumption and law of the conservation of energy, it is possible to affirm that after the same time interval the sum of all forms of energy of air in section I-I is equal to the sum of all forms of energy in section II-II, i.e.,

$$E_{\text{полн}_1} = E_{\text{полн}_2} = \text{const.}$$

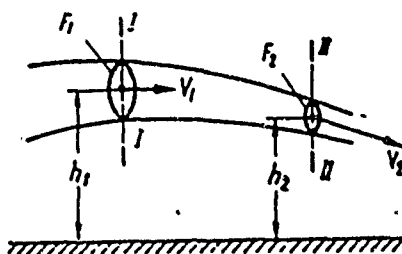


Fig. 2. Stream of ideal air under the condition when the exchange of energies between the stream and environment is absent.

The total energy ( $E_{\text{total}}$ ) is the sum of the kinetic and potential energy.

Under the condition that through sections I-I and II-II a mass of air of  $1 \text{ m}^3$  passes, the kinetic energy ( $E_{\text{kin}} = \frac{mV^2}{2}$ ) can be expressed as the product of density ( $\rho$ ) by the square of speed, since a mass of  $1 \text{ m}^3$  of air is its density.

The kinetic energy of  $1 \text{ m}^3$  of air ( $E_{\text{kin}} = \frac{\rho V^2}{2}$ ) is called dynamic pressure or impact pressure ( $q$ ), and it acts on the surface of a body immersed in the flow, in a direction perpendicular to streamlines, as is shown in Fig. 3 by the sign (+). Streamlines are called the trajectory of motion of particles of air of steady airflow.

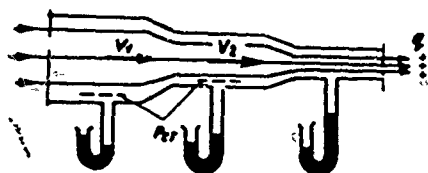


Fig. 3. Direction of action of static pressure and impact pressure:  $q$  - impact pressure (+);  $p_{\text{ст}}$  - static pressure (-).

Potential energy consists of the energy of forces of pressure, thermal energy and energy of the force of weight. Under the condition that the air is incompressible, between the stream and environment heat exchange is absent, and the energy of forces of weight is disregarded as an insignificance of it, then with horizontal flow of the stream when  $h_1 = h_2$  (see Fig. 2), the potential energy of  $1 \text{ m}^3$  air will be equal to the static pressure ( $p_{\text{ст}}$ ). Static pressure is called the pressure of layers of air on the surface of a body immersed into it parallel to the streamlines, as is shown in Fig. 3 by sign (-). Producing the appropriate replacement, the equation of constancy of energy can be written:

$$p_1 + \frac{\rho V_1^2}{2} = p_2 + \frac{\rho V_2^2}{2} = \text{const.}$$

This equation is called the equation of Bernoulli - the scientist, who applied the law of the conservation of energy to a stream of air (liquid).

From the Bernoulli equation one can draw the conclusion that the sum of static ( $p_{st}$ ) and dynamic ( $q = \frac{\rho V^2}{2}$ ) pressures in any section of the stream of steady flow of an ideal incompressible air is a constant magnitude.

The physical essence of the Bernoulli law consists of the fact that it establishes the connection between flow rate in the given section of the stream and pressure: the greater the airspeed, the lower the pressure and vice versa.

#### 6. Streamline Flow Pattern and Its Elements

With motion of a body in air, the latter tries to brake the motion of the body and, conversely, with the flowing around of a motionless body by airflow there appears a force which tries to shift the body from place and impart motion to it.

The force action of the airflow on a body is called aerodynamic force.

The magnitude of aerodynamic force does not depend on whether the body moves a certain speed in the airflow or the body is motionless, and on it there advances a flow of air with a speed equal to and opposite the speed of the body. This principle is called the reversibility of motion, and it is widely used for the carrying out investigations under laboratory conditions by means of wind tunnels.

The wind tunnel is an installation for the creation of airflow (Fig. 4).

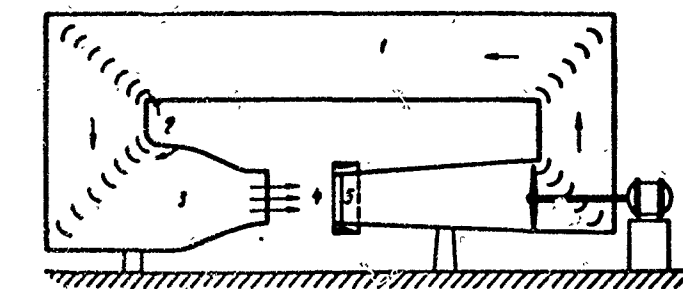


Fig. 4. Schematic diagram of a wind tunnel. 1 - return channel. 2 - turning elbows. 3 - collector. 4 - working section. 5 - diffuser.

The magnitude of aerodynamic force depends on the character of the flowing around of a body by air. The visible pattern of the streamline flow of a body by air is called the aerodynamic spectrum. The streamline flow pattern consists of the following elements (Fig. 5):

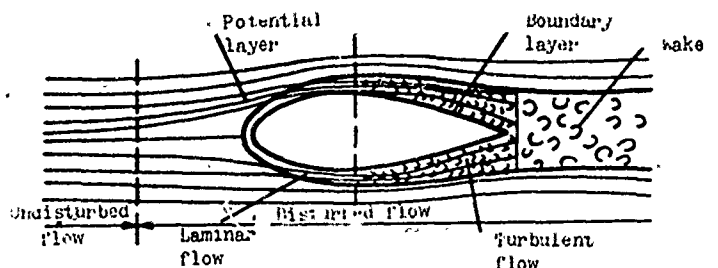


Fig. 5. Basic elements of the streamline flow pattern and forms of flows of air in the boundary layer.

1) undisturbed airflow the streams of which are not deformed by the body present in it;

2) disturbed airflow the streams of which are deformed by a present body in it.

The basic elements of disturbed airflow are the following.

1. Boundary layer - the layer of air in which the speed of particles is changed from zero to the speed of particles of an undisturbed flow. The flow of air in the boundary layer is subdivided into two forms:

1) laminar - with immiscible streams;

2) turbulent - with randomly vortex flow.

The formation of turbulent flow in the boundary layer is explained by the increase in speed of airflow with the removal of it from the surface of the body, as a result of which there appears a pair of forces, rotating the given volume of air (Fig. 6).

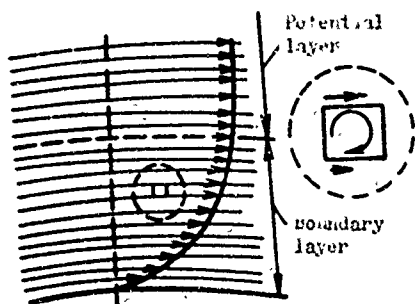


Fig. 6. Formation of turbulent flow in the boundary layer.

The point of transition from laminar flow to turbulent flow depends on the form of the body, degree of treatment of its surface and speed of the airflow. These factors and position of the body in the airflow are governed by the thickness of the boundary layer.

2. Potential layer - layer of air in which forces of internal friction do not appear.

3. Wake - vortex airflow trailing the streamlined body and moving behind it.

In their character the streamline flow patterns are subdivided into smooth and vortex, symmetric and asymmetric.

The streamline flow pattern of a flat plate located perpendicular to the flow is symmetric and vortex (Fig. 7).

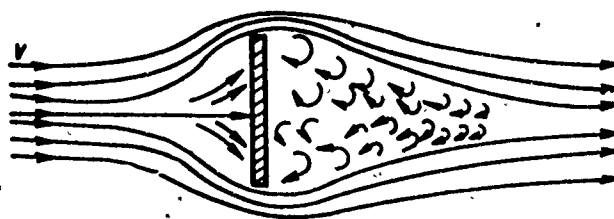


Fig. 7. Flow pattern of a flat plate.

The streamline flow pattern of a drop-shaped body, located so that its axis of symmetry coincides with the direction of the flow, is symmetric and smooth (Fig. 8).

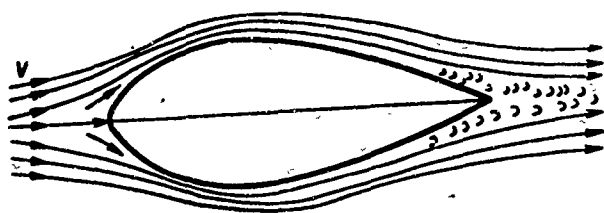


Fig. 8. Flow pattern of a drop-shaped body.

Bodies having a smooth flow pattern are called streamlined. Flow patterns of a flat plate, located at an angle to the incident flow, and of a symmetric streamlined body, also located at an angle to the incident flow, are asymmetric and vortex.

With flowing around of asymmetric bodies the patterns are asymmetric. With the help of a streamline flow pattern it is possible to establish where the streams become narrow and where they are expanded. Knowing this, on the basis of the law of constancy of flow rate of air per second, one can determine where the speed is greater and where it is less. Applying the Bernoulli law, it is possible to establish where the pressure is greater and where it is less and therefore judge the direction of the aerodynamic forces.

## 7. Basic Law of Air Resistance

By examining the streamline flow patterns of a flat plate and a drop-shaped body, it is possible to establish that due to deceleration in front of the body the flow rate decreases, and the pressure is increased. The degree of increase in pressure depends on the shape of nose cone of the body. In front of a flat plate the pressure is greater than it is in front of a drop-shaped body. Behind the body, due to rarefaction, the pressure decreases and for flat plate by a larger value as compared to that of a drop-shaped body.

Thus, in front of the body and behind it there will be formed a difference in pressures, as a result of which there is created an aerodynamic force which is called pressure drag.

Besides this, because of the friction of air in the boundary

layer there appears an aerodynamic force which is called friction drag.

With symmetric flowing around of a body the pressure drag and friction drag are directed to a side opposite the motion of the body and together comprise drag.

It is established by experiments that the aerodynamic force depends on flow rate, mass air density, shape and dimensions of the body, position in the flow and state of its surface.

With an increase in the approach stream velocity its kinetic energy, which is proportional to the square of the speed, is increased. Therefore, with the flowing around of a flat plate, directed perpendicular to the flow, with an increase in speed the pressure in the front section of it is increased, since the greater part of the kinetic energy of the flow with deceleration changes to potential energy of the pressure.

Behind the plate, with an increase in approach stream velocity, the pressure decreases even more, since because of the increase in inertness of the stream the extent of region of lowered pressure is increased. Thus, with an increase in flow rate because of the increase in the difference in pressure in front of the body and after it, the aerodynamic force is increased proportionally to the square of speed.

Earlier it was established that air density characterizes the inertness of it: the greater the density, the greater the inertness of the air. For the motion of a body in more inert and, consequently, denser air, it is required to apply greater forces for the shift of particles of the air, and this means that the air will act with greater force on the body. Consequently, the greater the air density, the greater the aerodynamic force acting on the moving body.

In accordance with the laws of mechanics, the magnitude of aerodynamic force is proportional to the area of the section of the body perpendicular to the direction of action of the given force.

For the majority of bodies such a section is the biggest cross section, called mid-section, and for the wing - the area of it in the plan.

The form of the body affects the character of the streamline flow pattern (velocity of streams flowing around the given body) and, consequently, the difference in pressures which determines the magnitude of the aerodynamic force. With a change in position of the body in the airflow the flow pattern of it is changed, which involves a change in magnitude and direction of the aerodynamic forces.

Bodies having less rough surface experience less frictional forces, since on a greater surface their boundary layer has a laminar flow, and in laminar flow of the boundary layer the friction drag is less than that in a turbulent.

Thus, if the effect of the form and position of the body in the flow and state of treatment of its surface are considered by the correction factor, which is called the aerodynamic coefficient, then the conclusion can be made that the aerodynamic force is directly proportional to its coefficient, impact pressure and area of mid-section of the body (for the wing - its area).

If one were to designate the full aerodynamic resisting force of air by the letter  $R$ , the aerodynamic coefficient of it by  $c_R$ , impact pressure -  $q$  and wing area -  $S$ , then the formula of the resisting force of air can be thus written:

$$R = c_R q S,$$

and since the impact pressure is equal to  $\rho V^2/2$ , the formula will have the form:

$$R = c_R \frac{\rho V^2}{2} S.$$



The given formula of the resisting force of air is the basic one, since according to formulas analogous to it one can determine the magnitude of any aerodynamic force by replacing only the designation of force and its coefficient.

## CHAPTER II

### AERODYNAMIC FORCES OF WING

The wing is the main part of an aircraft and is designed for the creation of lift, supporting the aircraft in air.

#### 1. Geometric Characteristics of the Wing

The magnitude and direction of aerodynamic forces acting on the wing is determined by the form of profile of the wing, contour of the wing in the plan, front view of the wing, which comprise the geometric characteristics of the wing.

The profile of the wing is called the form of cross section of the wing. With respect to form the profiles wings are symmetric and asymmetric. Symmetric profiles pertain to biconvex profiles, and asymmetric profiles pertain to biconvex, planoconvex, concavoconvex, and S-shaped.

The profile of the wing of the An-2 aircraft is asymmetric and biconvex (Fig. 9).

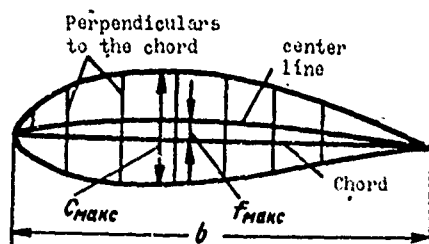


Fig. 9. Airfoil camber.

The basic parameters of the profile of a wing are length of the chord ( $b$ ), relative thickness ( $\bar{c}$ ) and relative curvature ( $\bar{f}$ ).

The airfoil chord of a wing is called the segment of a straight line connecting extreme points of the leading edge and trailing edge (measured in meters).

The thickness ratio of the profile of a wing ( $c$ ) is called the ratio of the maximum thickness ( $c_{max}$ ) to the length of the chord expressed in percent.

$$\bar{c} = \frac{c_{max}}{b} \cdot 100.$$

The thickness ratio of profiles of contemporary wings amounts to 8-16%. Profiles having a thickness ratio of up to 8% are thin, from 8 to 12% - medium and over 12% - thick. The thickness ratio of the profile of a wing of the An-2 aircraft is 14%.

The relative airfoil camber ( $f$ ) is called the ratio of maximum curvature ( $f_{max}$ ) to the chord expressed in percent:

$$\bar{f} = \frac{f_{max}}{b} \cdot 100.$$

The airfoil camber is called the distance between the chord and center line of the profile. The center line of the profile of the wing is called the line connecting the middles of segments of the profile perpendicular to the chord (see Fig. 9). An increase in airfoil thickness and airfoil camber up to definite magnitudes and displacement forward, toward the leading edge, of the maximum thickness and maximum curvature increase the carrying properties of the wing but simultaneously cause an increase in drag.

Wings with a thick profile and relative curvature up to 2% are used on aircraft designed for relatively low speeds of flight.

According to the planform wings are rectangular, trapezoidal, ellipse-shaped and swept-back. The best aerodynamic properties are

possessed by ellipse-shaped wings however, because of the complexity of production, they are not widespread. The rectangular wing, used on certain aircraft, has the worst aerodynamic characteristics but possesses these advantages: simplicity of manufacture and the best stability at combat angles of attack, since its flow separation starts not on the cantilevers but in the root.

The An-2 aircraft has a biplane cell of wings, which consists of an upper and lower wing, band braces and biplane struts. The upper and lower wings have an identical rectangular form with rounded cantilevers.

The form of the wing is characterized by span, area and aspect ratio. Wing span ( $l$ ) is called the distance between extreme points of cantilevers of the wing in a direction perpendicular to the longitudinal axis of the aircraft. The span of the upper wing of the An-2 aircraft is 18 m and the span of the lower wing - 14 m.

The wing area ( $S$ ) is called the area limited by the contour of it, which is determined by the formula

$$S = lb.$$

The total area of the wing cell of the An-2 aircraft is  $17.5 \text{ m}^2$ .

The wing aspect ratio ( $\lambda$ ) is called the ratio of the wing span to the chord  $\lambda = l/b$ . The wing aspect ratio has a considerable influence on the magnitude of induced drag and lateral stability of the aircraft. With an increase in wing aspect ratio the induced drag decreases, and lateral stability of the aircraft is improved. The wing aspect ratio of contemporary transport aircraft is within 6-10 and is regulated by the strength; with an increase in speed of flight it has a tendency to decrease. The wing aspect ratio of the An-2 aircraft is upper - 7.7 and lower - 7.25.

The form of the wing is characterized by the dihedral angle ( $\psi$ ). The dihedral angle of the wing is called the angle included between

the lateral axis of the aircraft and lower outline of the wing. The dihedral angle of the An-2 aircraft is for the lower wing -  $4^{\circ}19'$  and upper wing -  $3^{\circ}$ .

To improve flight characteristics of an aircraft wings with a geometric or aerodynamic twist over the span are sometimes used. The geometrically twisted wing is called the wing whose chords are not in one plane. The aerodynamically twisted wing is called such a wing which is gathered from different profiles. The wing of the An-2 aircraft has a profile constant in span.

## 2. Total Aerodynamic Force and Its Components

When the wing encounters airflow, inasmuch as the curvature of it from above is greater than that from below, according to the law of constancy of the flow rate of air per second, the local velocity of the flowing around it above is greater than that below, and for the leading edge it will decrease sharply and at separate points drop to 0. According to the law of Bernoulli, in front of the wing under it there will appear a region of raised pressure and above the wing and behind it - lowered pressure. Furthermore, due to the viscosity of the air there appears a frictional force in the boundary layer. The pattern of distribution of pressure along the profile of the wing depends on the position of the wing in the airflow, for orientation of which the concept on angle of attack is used.

The angle of attack of the wing ( $\alpha$ ) is called the angle between the direction of the chord of the wing and incident flow of air or direction of the velocity vector of flight (Fig. 10).

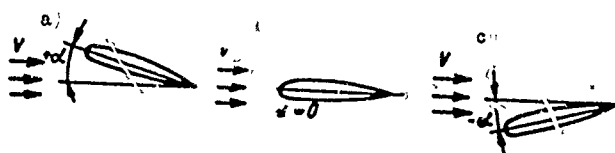


Fig. 10. Angle of attack of the wing: a)  $\alpha > 0$ ; b)  $\alpha = 0$ ; c)  $\alpha < 0$ .

To obtain the pattern of the distribution of pressure along the profile of the wing a battery manometer and drained model of the wing. The drained model of the wing is called a model of the wing with drillings, to which hoses from the battery manometer are joined. The distribution of pressure along the profile is shown in the form of vector diagrams. For the construction of a vector diagram we trace the profile of the wing, mark on it points at which pressure was measured, and from these points plot by vectors magnitudes of excess pressures. Since the vectors characterize pressure, they are plotted perpendicular to the contour of the profile. If at a given point the pressure is lowered, the vector is directed by a arrow from the profile; if, however, the pressure is raised, then the vector is directed to the profile. Ends of vectors are connected by common line and obtain a vector diagram of the distribution of pressures.

Figure 11a shows the pattern of distribution of pressures along the profile of the wing at small angles of attack, from which it is clear that the greatest rarefaction is obtained on the upper surface of the wing at the place of the greatest narrowing of the streams. At an angle of attack equal to zero, the greatest rarefaction will be at the place of the largest thickness of the profile. Under the wing there also occurs a narrowing of the streams, as a result of which a zone of rarefaction will be there, but less than that above the wing. In front of the nose of the wing there is a region of raised pressure.

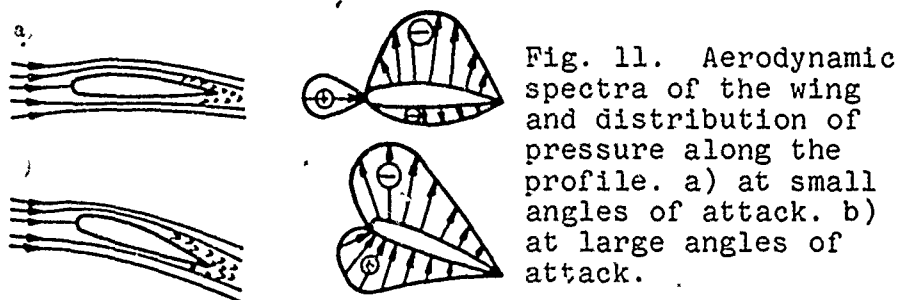


Figure 11b shows the pattern of distribution of pressure along the profile of the wing at a large angle of attack. With an increase

in angle of attack the zone of rarefaction is displaced toward the leading edge and is considerably increased in magnitude. This occurs because the place of the greatest narrowing of the streams moves toward the leading edge. Under the wing particles of air, encountering the lower surface of the wing, are braked, as a result of which the pressure is increased.

Every vector of excess pressure shown on the diagram constitutes a force acting on a unit of surface of the wing. By summing all the vectors, there can be obtained an aerodynamic force, neglecting forces of friction. The given aerodynamic force, taking into account the frictional force of air in the boundary layer, will comprise the total aerodynamic force of the wing. Thus, the total aerodynamic force (R) appears because of the difference in pressures in front of the wing and behind it, under the wing and above it, and also as a result of the friction of air in the boundary layer.

The point of application of total aerodynamic force is on the chord of the wing and is called the center of pressure (CP). Inasmuch as the total aerodynamic force acts in the direction of lower pressure, it will be directed upwards and deflected back (Fig. 12).

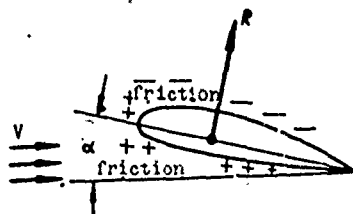


Fig. 12. Appearance of total aerodynamic force of the wing.

According to the basic law of air resistance the total aerodynamic force can be defined by the formula

$$R = c_R \frac{\rho V^2}{2} S.$$

By experimental means it has been established that for a wing of an asymmetric profile, with an increase in angle of attack the total aerodynamic force is increased, and the center of pressure is displaced forward toward the leading edge.

Figure 13 shows that with the decomposition of total aerodynamic force there can be obtained the following:

- a) in a direction perpendicular to the undisturbed flow, lift ( $Y$ );
- b) in the direction of the airflow, in an opposite direction to the motion of the wing, drag ( $Q$ ).

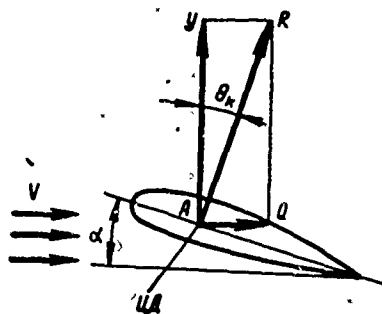


Fig. 13. Decomposition of total aerodynamic force into components.

The angle between vectors of lift and total aerodynamic force is called the angle of efficiency ( $\theta_k$ ).

### 3. Lifting Force of the Wing

With the flow around of biconvex asymmetric profile of the wing, under the wing the speed is greater and the pressure less than they are in an undisturbed flow; above the wing the speed is greater and the pressure less than those under the wing. Consequently, lift appears because of the difference in pressures under the wing and above the wing basically because of the rarefaction above the wing. The magnitude of lift is determined by the formula

$$Y = c_y \frac{\rho V^2}{2} S,$$



where  $c_y$  - coefficient of lift.

The coefficient of lift is determined by experimental means with testing of the wing in a wind tunnel and depends on the angle of attack, form of the profile of the wing, planform of the wing, and degree of treatment of the surface of the wing. The coefficient of lift characterizes the carrying capacity of the wing. For the given wing the coefficient of lift depends only on the angle of attack.

If according to data of testing a wing of asymmetric profile in a wind tunnel at different angles of attack we construct a graph, then it will look as is shown on Fig. 14.

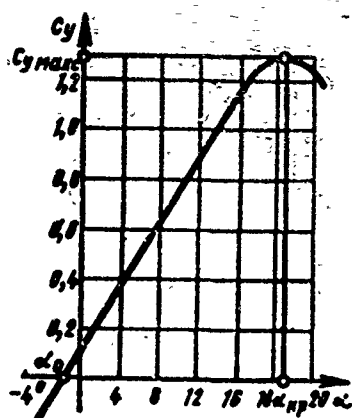


Fig. 14. Graph of the dependence of the coefficient of lift ( $c_y$ ) of a wing with an asymmetric profile on the angle of attack ( $\alpha$ ).

From the figure one can see the following:

- 1) at a certain negative value of the angle of attack the coefficient of lift is equal to zero. This angle of attack is called the angle of attack of zero lift and is designated  $\alpha_0$ ;
- 2) with an increase in angles of attack up to a certain value the coefficient of lift is increased proportionally (along a straight line); after a certain value of the angle of attack the increase in the coefficient of lift decreases, which is explained by the formation of eddies on the upper surface of the wing;
- 3) at a definite value of the angle of attack the coefficient of lift reaches a maximum value, and then with a further increase

in the angle of attack it decreases. The angle of attack at which the coefficient of lift reaches a maximum value is called critical and is designated by  $\alpha_{kp}$ .

The decrease in the coefficient of lift at angles of attack greater than the critical, as is shown in Fig. 15, occurs because of the intense flow separation from the wing induced by the motion of the boundary layer against the motion of the basic flow.



Fig. 15. Flow separation at angles of attack beyond stalling: at point A the pressure is greater than at point B, and at point C the pressure is greater than in points A and B (the arrow shows the direction of motion of the boundary layer).

Angles of attack - from the angle of attack of zero lift up to the angle of critical attack - comprise the range of flying angles of attack. However, the flight of an aircraft at angles of attack close to the critical is not produced, since at these angles of attack the aircraft does not possess sufficient stability and are difficult to control. There is practical interest in angles of attack at which the coefficient of lift is changed in proportional to the change of angles.

#### 4. Profile Drag of a Wing

The drag of a wing formed because of the difference in pressures in front of the wing and behind it is called pressure drag. The drag of a wing induced by friction of particles of air in the boundary layer is called by friction drag.

Pressure drag and friction drag together comprise the profile drag. The magnitude of the profile drag is determined by the formula

$$Q_p = c_{x_p} \frac{\rho V^2}{2} S,$$

where  $c_{x_p}$  - coefficient of profile drag of the wing.

The magnitude of pressure drag depends on the thickness ratio of the airfoil and relative airfoil camber, with an increase of which it is increased.

At small angles of attack the pressure drag is changed insignificantly. The magnitude of friction drag depends only on degree of treatment of the surface of the wing and comprises about 80% of the whole profile drag.

#### 5. Induced Drag of a Wing

Before 1910 scientists considered that the drag of a wing appeared only because of the difference in pressures in front of and behind the wing and friction of the air in the boundary layer. In 1910 scientist S. A. Chaplygin, investigating the flow around of a wing, established that drag depends also on the difference in pressures under the wing and above it. In the presence of a difference in pressures masses of air will overflow through cantilevers of the wing from the region of raised pressure into the region of lowered pressure - onto the wing, as a result of which end vortexes will be formed (Fig. 16). End vortexes, being directed from the bottom to the top, cause in the region of the wing a lowering of the whole flow downwards, which leads to downwash.

Inasmuch as under the action of vertical velocity ( $V_y$ ) there occurred downwash under the wing, then the lift of the wing will be deflected and will act in a direction perpendicular to the true direction of the flow, as was shown in Fig. 17.

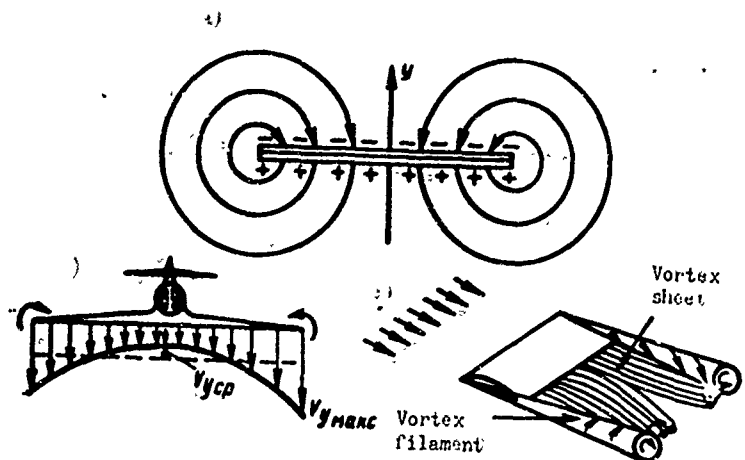


Fig. 16. Flowing around of a finite span. a) overflowing of airflow; b) distribution of vertical velocity along the span; c) formation of vortex filaments and a vortex sheet.

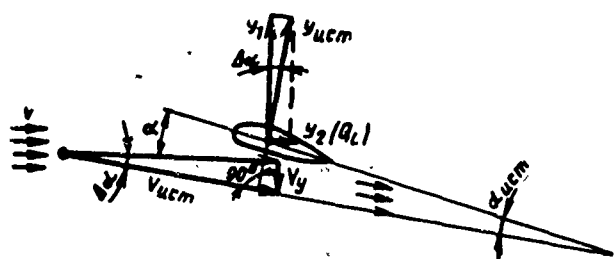


Fig. 17. Formation of induced drag.

Decomposing the true lift into two directions - perpendicular to the undisturbed flow ( $Y_1$ ) and in direction of the airflow ( $Y_2$ ), we are convinced of the fact that the horizontal projection of the true lift coincides with the direction of action of the resisting force and increases it.

The horizontal projection of true lift ( $Y_2$ ) is induced drag ( $Q_1$ ). It appeared because of downwash under the wing with the formation of lift and is determined by the formula

$$Q_1 = c_{x_i} \frac{\rho V^2}{2} S,$$

where  $c_{x_i}$  - coefficient of induced drag, which depends on the coefficient of lift ( $c_y$ ), the planform of the wing, which is considered the coefficient  $A$ , and wing aspect ratio ( $\lambda$ ). The coefficient of induced drag is determined by the formula

$$c_{x_i} = A \frac{c_y^2}{\lambda}.$$

From the formula it is clear that with an increase in angle of attack the induced drag is increased, and with an increase in wing aspect ratio it decreases. Investigations showed that the greatest downwash under a wing is created by a wing of rectangular form, in view of which the induced drag of a wing of rectangular form is considerably greater than that for a wing of another form.

## 6. Drag of a Wing

The sum of profile and induced drags comprises the total drag or drag of the wing and is determined by the formula

$$Q = c_x \frac{\rho V^2}{2} S,$$

where  $c_x$  - coefficient of drag, dependent on the angle of attack, form of the profile planform of the wing and degree of treatment of the surface of the wing.

The graph plotted according to data of testing a wing in a wind tunnel at different angles of attack is shown in Fig. 18.

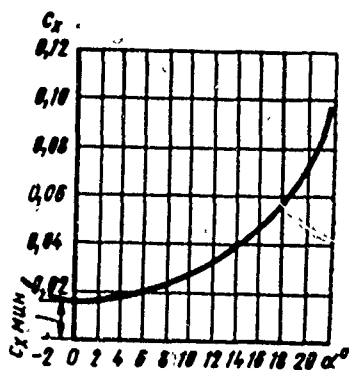


Fig. 18. Graph of the dependence of coefficients  $c_x$  on the angle of attack of the wing  $\alpha$  with an asymmetric profile.

From the figure one can see:

1) the coefficient of drag at none of the angles of attack is not equal to zero, since the flow around of the profile cannot occur without drag;

2) at an angle of attack close to the angle of attack of zero lift, the coefficient of drag has a minimum value; this angle is called the angle of attack of minimum drag and is designated  $\alpha_{c_{x_{min}}}$ ;

3) with a change in angles of attack on both sides of the angle of attack of minimum drag, the coefficient of drag is increased: but since the coefficient of profile drag in the range of flying angles of attack is changed insignificantly, and the coefficient of induced drag is proportional to the square of the coefficient of lift, then an increase in coefficient of drag occurs basically because of the increase in induced drag;

4) with an approach to the critical angle of attack the increase in the coefficient of drag is considerably increased because of the intense separation of flow and at an angle of attack equal  $90^\circ$  it will reach a maximum value, since the flow pattern of the wing is analogous to the flow pattern of a flat plate.

## 7. Lift-Drag Ratio of a Wing

The number showing how many times the lift of a wing at a given angle of attack is greater than drag is called the lift-drag ratio:

$$K = \frac{V}{Q} = \frac{c_y}{c_x}.$$

The magnitude of the lift-drag ratio is the means of judging the aerodynamic perfection of the wing. Inasmuch as coefficients of lift and drag depend on the angle of attack, the lift-drag ratio depends on the angle of attack. The angle at which the ratio reaches a maximum value is called the most advantageous and is designated by  $\alpha_{HB}$ .

With an increase in the angle of attack up to the most advantageous the ratio is increased, since an increase in lift progresses over an increase in drag, and with further increase in the angle of attack the ratio decreases. Between the ratio and angle of efficiency exists an inverse relation  $\theta_x = \frac{1}{K}$ . Quantity  $\frac{1}{K}$  is called inverse efficiency.

## 8. Polar of a Wing

The numerical values of coefficients of lift and drag are determined by experimental means with a test of a wing in a wind tunnel. A model of a wing is set in the working section of the pipe on wind tunnel balances at an assigned angle of attack, and then the fan is started. With the flowing around of the model of the wing by an airflow exerted on it are effects of aerodynamic forces, the magnitude of which is taken from readings of the wind-tunnel balances. Having measured the magnitudes of lift and drag, one determines the value of coefficients:

$$c_y = \frac{Y}{qS}; \quad c_x = \frac{Q}{qS}.$$

Compiling a table of values of aerodynamic coefficients at different angles of attack from the angle of attack of zero lift up to angles of attack beyond stalling it is possible to show visually their dependence of coefficients of lift and drag of the wing at different angles of attack is called the polar of the wing.

For the construction of a polar axes of coordinates are used: plotted along the axis of the abscissas is the value  $c_x$  and along the axis of the ordinates -  $c_y$ .

A graph is called a polar because it shows the locus of ends of vectors of coefficients of the total aerodynamic force, and it can be examined as a polar diagram in coordinates  $c_R$  and  $\phi$ , where  $\phi$  is the angle of inclination of the total aerodynamic force or its coefficient to the direction of speed of the airflow.

Figure 19 shows that if one were to draw from pole "0", which is the center of pressure, a vector to any angle of attack of the polar, then it will constitute a diagonal of a rectangle, the sides of which are respectively equal to  $c_y$  and  $c_x$  and the angle between  $c_y$  and  $c_R$  - angle of efficiency ( $\theta_R$ ). Since the coefficient of drag is several times less than the coefficient of lift, then the polar plotted in identical scales has in the range of flying angles of attack an insignificant curvature, which hampers the exact determination of value  $c_x$  with respect to it. Proceeding from this, in many cases with the construction the polars of  $c_x$  are plotted in a scale five and ten times larger than  $c_y$ . The use of such a polar is analogous, with the exception of the fact that it is impossible to determine by it  $c_R$  and  $\theta_R$  values.

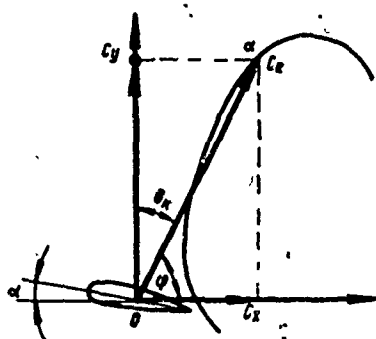


Fig. 19. Essence of the polar.

On the polar one can determine the following characteristic angles (Fig. 20):

- 1) angle of attack of zero lift ( $\alpha_0$ ) - crossing of the polar with the axis  $c_x$ ;
- 2) angle of attack at which  $c_x$  has a minimum value ( $\alpha_{c_{x_{min}}}$ ) - at the point of contact of the tangent to the polar drawn in parallel to the axis of the ordinates.
- 3) the angle of attack is most advantageous at which the efficiency is maximum, and the angle of efficiency is minimum ( $\alpha_{\theta_R}$ ) at the point of contact of the tangent to the polar drawn from the



origin of the coordinates;

4) the angle of attack critical at which the coefficient of lift reaches a maximum value ( $\alpha_{kp.}$ ) at the point of contact of the tangent to the polar drawn in parallel to the axis of the abscissas;

5) two different angles of attack at an identical angle of efficiency - secant from the origin of the coordinates.

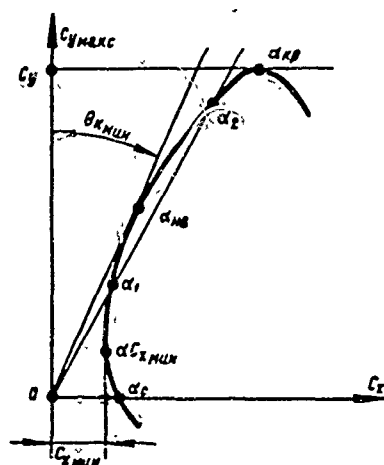


Fig. 20. Characteristic angles of attack on polar of a wing.

#### 9. Methods of Increasing the Maximum Lifting Force of a Wing

In order to create lift equal to the weight of the aircraft at minimum flight speed, it is necessary to increase the thickness and airfoil camber, which leads to an increase in drag and considerable decrease in maximum speed of flight of the aircraft. For the creation of lift, which balances the flying weight of an aircraft, at a comparatively low speed a wing equipped with means of mechanization is used. The theory of the mechanized wing was first developed by S. A. Chaplygin.

The applicable means of mechanization of a wing are subdivided into three forms: shields, flaps and slats. All these means of mechanization are designed for the improvement of takeoff and

landing characteristics of the aircraft. On the An-2 aircraft slot flaps and automatic sliding slats are used.

Flaps are a profiled surface, located in the trailing section of the wing, are capable of turning around the axis passing through the nose of the flap. The angle of deflection of the flaps at takeoff is up to  $30^\circ$  and during landing - up to  $40^\circ$ .

An increase in maximum value of the coefficient of lift ( $c_y$ ) with the deflection of flaps is reached basically owing to the increase in the airfoil camber. With the deflection of the flaps at a small angle and at comparatively low flight speeds the value  $c_x$  in practice almost does not change; which with an increase in  $c_y$  promotes an increase in efficiency. Deflection of the flaps at an angle up to  $5^\circ$  with the flight speed of the An-2 aircraft up to 150 km/h is accompanied by an increase in speed of horizontal flight of 5 km/h and vertical climb rate - 0.2 m/s. With deflection of the flaps at a large angle and at comparatively high speeds of flight,  $c_x$ , conversely, is increased considerably more than  $c_y$ , as a result of which the efficiency of the aircraft decreases.

The presence of a slot flap gives an additional increase in  $c_y$ , since the air passing through the profiled slot (Fig. 21) with greater speed blows off the eddy from the upper surface of the wing at the trailing edge and decreases the thickness of the boundary layer. With deflection of the flap the conditions of flow around of the wing at the trailing edge are improved, and the center of pressure is displaced back.

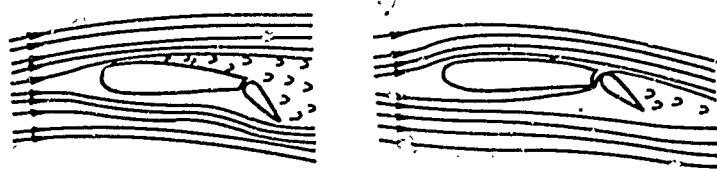


Fig. 21. Principle of action of a slot flap. a) simple flap; b) slot flap.

The slat is a small profiled wing located on the leading edge of the wing. With flight of the aircraft at small angles of attack the slat is closely pressed to the wing, being inscribed into the general contour of it. At large angles of attack, under the action of aerodynamic forces the slat is automatically released, forming a slot through which airflow rushes (Fig. 22). With the motion of airflow through the slot the speed of it is increased, part of the eddies on the upper surface of the wing is blown off, and the thickness of the boundary layer decreases, as a result of which  $c_y$  is increased. Simultaneously with the release of the slat the pressure in front of the wing decreases,  $c_x$  decreases, and the efficiency is increased.

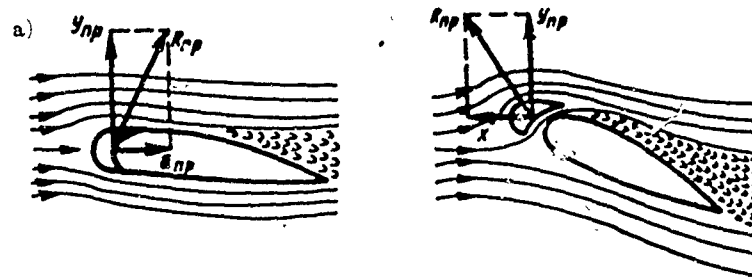


Fig. 22. Streamline flow pattern of a wing and aerodynamic forces of a slat. a) at small angles of attack (slat is pressed to the wing); b) on large angles of attack (slat is released from the wing).

The slats also have considerable deficiencies:

1) since it is impossible to achieve the exact trimming of the slat over the whole wing span, the presence of the slot during flight of the aircraft at small angles of attack decreases the difference in pressures under the wing and above it, as a result of which for the creation of lift high speed of flight is required;

2) during takeoff with a cross wind because of asymmetric flowing around of the wing cell one-sided departure of the slat can occur, which considerably complicates the technology of fulfillment of take-off and lowers the flight safety.

### CHAPTER III

#### PECULIARITIES OF ARRANGEMENT AND AERODYNAMIC CHARACTERISTICS OF THE An-2 AIRCRAFT

##### 1. Aerodynamic Forces of an Aircraft

The lift of an aircraft is equal to the lift of the wing, since lift created by remaining elements of the construction of the aircraft is so insignificant that it is practically disregarded.

The drag of an aircraft consists of the drag of its separate parts streamlined by airflow:  $Q$  of the aircraft =  $Q$  of the wing +  $Q$  of the fuselage +  $Q$  of the empennage +  $Q$  of the propulsion system +  $Q$  of the landing gear +  $Q$  of the special equipment and others.

The drag of all noncarrying parts of an aircraft is called parasite drag and is designated by  $Q_{bp}$ . In the range of flying angles of attack coefficient  $c_{x_{bp}}$  is changed insignificantly, and therefore its magnitude is considered constant, not depending on angles of attack.

##### 2. Lift-Drag Ratio of an Aircraft

The lift-drag ratio of an aircraft is the number showing how many times the lift at a given angle of attack is greater than the drag of an aircraft. Inasmuch as the lifting force of an aircraft is equal to the lift of a wing, and the drag of an aircraft is greater than the drag of a wing by the magnitude of the parasite drag, then the efficiency of the aircraft is always less than the efficiency of the wing.

The efficiency of the aircraft, just also the efficiency of the wing, depends on the angle of attack: with an increase in the angle of attack up to the optimum the efficiency is increased, and with a further increase in the angle of attack the efficiency decreases. Besides the angle of attack the efficiency of the aircraft is affected by interference, the state of the surface of external (especially carrying) parts of the aircraft and slip.

Interference is called the mutual effect of articulated parts of the aircraft on the magnitude of its aerodynamic forces. As a result of the interference, as a rule, drag is increased, and lift decreases, which leads to a decrease in the lift-drag ratio. The phenomenon of interference appears at places of the joint of the wing with the fuselage (on the An-2 aircraft, especially at the joint of the lower wing with the fuselage), fuselage with the empennage, and agricultural equipment with the fuselage. The reason of the appearance of interference is the formation of the diffusion effect. Thus, for example, the lateral surface of the fuselage and surface of the wing will form a channel expanded in the direction of motion of the airflow - a diffuser, which is shown in Fig. 23. In the diffuser with expansion of the channel, the speed of the airflow decreases, and pressure is increased.

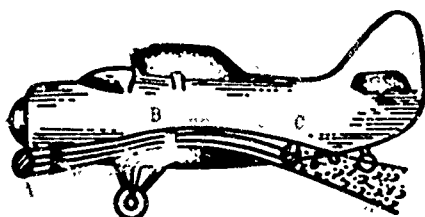


Fig. 23. Phenomenon of interference (A, B, C - different sections of the stream).

The formed difference in pressures (in section B - less, in section C - greater) causes motion of the boundary layer toward the basic flow, which leads to swelling of the boundary layer, the lag of it from the surface of the wing and separation from the wing by the basic flow. Local separation of the boundary layer involves a decrease in lift and increase in drag.

To decrease the interference fairings are used; these provide smoothness of transition at places of the joint of parts of the aircraft with the fuselage.

An increase in roughness of the surface and distortion of the form of the profile of elements of construction of the aircraft, induced by the presence of dirt, the formation of waviness of the skin, icing, and also by the disturbance of hermetic sealing of the aircraft because of the loose adjoining of operational hatches and fairings, decrease the lift-drag ratio of the aircraft.

With rudder deflection or under the influence of a cross wind a moment is created which tries to change the direction of flight of an aircraft, and since the aircraft by inertia tries to move rectilinearly, a slanting motion of the aircraft, which is called slip is obtained. The angle formed by the plane of symmetry of the aircraft and direction of the speed of flight is called the angle of side slip and is designated by the letter  $\beta$ . Figure 24 shows that in the presence of slip in the creation of lift there participates only the component of speed  $V_1$ , in view of which the lift decreases. On the other hand, due to the asymmetric flowing around of the fuselage and other parts of the aircraft, the coefficient of drag is increased. As a result of a decrease in lift and increase in drag, during flight of an aircraft with slip the lift-drag ratio of it decreases.

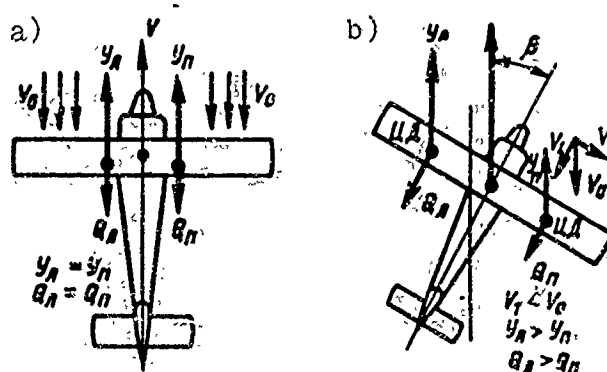


Fig. 24. Influence of slip on lift-drag ratio of an aircraft: a) flight of an aircraft without slip; b) flight of an aircraft with slip.

### 3. General Characteristic and Aerodynamic Peculiarities of the Arrangement of the An-2 Aircraft

#### General Characteristic

The An-2 aircraft, developed by the Special Design Office (OKB) under the leadership of the general designer, Hero of Socialist Labor, O. K. Antonov, was made operational as a multipurpose aircraft. It is used on feeder airlines as a transport aircraft for transporting passengers and cargoes, and with insignificant re-equipment it is used in other variants:

- 1) agricultural - for combatting pests of agriculture, aerial sowing and the fertilizing of sowings;
- 2) medical - for use in medical aviation;
- 3) water - for the exploitation on river routes and in the Transarctic region - on floats, designated the An-2V;
- 4) for geological searches;
- 5) for putting out forest fires.

The aircraft has about 20 modifications and is used for the execution of over 30 different forms of works in the national economy of our country, and also far beyond its limits.

The normal flying weight for all versions of the aircraft is 5250 kg and in Alpine regions - 5000 kg.

It is permitted to transport the following on the aircraft: in the land version - 12 passengers; in the water version - 9 passengers.

In the cargo version on the aircraft there can be transported various kinds of cargo weighing up to 1200 kg and in the agricultural

version - up to 1370 kg of chemical poisons. The strength of the floor ensures transportation of concentrated cargoes with a load up to 1000 kgf/m<sup>2</sup>. In the winter period of the year, with sufficient thickness of snow cover, the aircraft operates on a ski landing gear.

The aircraft is equipped with modern flight equipment, allowing operation of it in any climatic and meteorological conditions both by day and at night. The flight crew consists of a commander of the crew (he is the chief pilot) and a copilot.

The aircraft possesses high flight characteristics. The range of indicated cruising speeds of the aircraft in horizontal flight is 145-225 km/h. A distinctive peculiarity of the An-2 aircraft, as compared to other aircraft, is the combination a comparatively long flying range and load capacity with good take off and landing data, which provide operation of it from field airfields and landing strips of limited dimensions, which is especially important in the fulfillment of operations in agriculture.

#### Peculiarities of the Arrangement

The aircraft is a biplane of the braced one-strut type with a nonretractable tricycle landing gear with a tail wheel (Fig. 25). As compared to the monoplane the biplane has an advantage in the fact that it creates lift of necessary magnitude at comparatively less flight speed of the aircraft, since it has a large wing area. A deficiency of the biplane consists in the fact that it creates great drag.

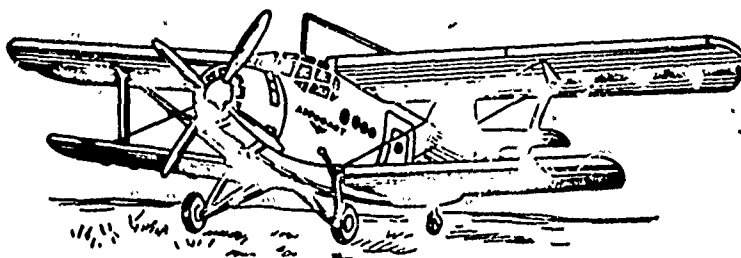


Fig. 25. General view of the An-2 aircraft.



A fuselage of streamlined form with a smooth boss in the nose part from above formed by the canopy of the pilots.

On each side the canopy emerges outside the dimensions of the fuselage, and this ensures good view to the sides and partially back. The nose part of the fuselage is made in the form of a truncated cone and the cross section of the main part - trapezoidal form with a narrowing of the top and curvature of angles and sides. In the central part from below the fuselage has smooth transitions to the center section, to which removable parts of the lower wing are joined. Toward the tail part the cross section of the fuselage decreases and passes to an oval form.

Drag of the fuselage comprises about 30% of the total drag of the aircraft and depends on the position of it in the airflow. To decrease drag of the fuselage, the wing is braced to it at a certain setting angle.

The setting angle is the angle formed by the longitudinal axis of the aircraft and root chord of the wing. The setting angle of the upper wing is  $3^\circ$  and of the lower wing -  $1^\circ$ . The magnitude of the setting angle is selected with such calculation so that with horizontal flight of the aircraft at angles of attack of  $1.5-4^\circ$  the angle between the axis of symmetry of the aircraft and direction of the airflow is small that drag of the fuselage considerably decreases.

The difference in magnitude of the setting angles of the upper and lower wing is caused by different conditions of their operation; the lower wing operates in the worse conditions than does the upper, and consequently, the separation of flow on the upper wing could come later than that on the lower. With such a difference in the magnitude of setting angles, flow separation on the upper and lower wings occurs simultaneously.

In the nose part of the fuselage there is placed the propulsion system, which consists in a radial reciprocating engine ASh-62IR and propeller of variable pitch (VISH). On the aircraft of transport

of the ailerons there is an improvement in lateral controllability of the aircraft at large angles of attack.

The ailerons have a distinctive peculiarity, which consists in the fact that with deflection of the flaps at an angle of  $40^\circ$  the ailerons are simultaneously deflected downwards at an angle of  $16^\circ$  and work as flaps, supplementing by this the positive action of the main flaps. However, in this case the critical angles of deflection of the ailerons as devices of lateral controllability of the aircraft are considerably changed, and if one were to consider from the chord of the wing, the angle of deflection of them will be: upwards -  $12^\circ$ ; downwards -  $30^\circ$ .

The empennage is of high location with a stabilizer of the semi-cantilever type. The high location of the empennage decreases the influence of airflow running off from the wing, which improves the longitudinal stability and controllability of the aircraft.

The vertical fin and rudder, stabilizer and elevator have a general streamlined biconvex symmetric profile. The vertical fin has a trapezoidal form with a curvature of whole contour along the leading edge and smooth transition of the lower part of the leading edge to the fuselage, which decreases drag from interference. The rudder has a smooth oval contour along the trailing edge. The large area of the vertical empennage ( $5,85 \text{ m}^2$ ) with clamped control provides good directional stability of the aircraft, and the large angle of deflection of the rudder ( $28^\circ$  to each side) in combination with the large area of the vertical empennage provides sufficient directional control of the aircraft during landing, when the speed is comparatively low and blowoff of the rudder by airflow rejected by the propeller is absent.

The stabilizer has in the plan a rectangular form with rounded ends; a profile of somewhat decreasing thickness is on the tail section. Relative to the construction horizontal line of the aircraft the stabilizer has a negative setting angle equal to  $1^\circ$ , which provides longitudinal equilibrium of the aircraft with insignificant elevator deflection during flight in basic cruise conditions.

On aircraft above the '60th series the elevator angle deflection is  $22^\circ$  downwards and  $42^\circ$  upwards. A large elevator angle deflection upwards provide creation for the aircraft of a landing angle of attack at low speeds and limiting nose-heaviness. The elevator angle deflection downwards ensures overcoming the positive pitching moment in takeoff conditions with deflected flaps and limiting tail-heaviness. To decrease the forces on the aircraft control vane there are aerodynamic axial compensation and also tabs set on the left aileron with deflection up and down of  $24^\circ$ , on the left half of the elevator with a deflection up and down of  $14^\circ$ , on the rudder with deflection to both sides of  $14^\circ$ . To eliminate vibrations the ailerons and rudder have 100% weight balancing and the elevator - 105%.

To decrease drag of the landing gear, the shock struts of the main wheels and upper attachment points struts to the truss of the center section and also parts of installation of the tail wheels protruding from the fuselage are closed by fairings.

The application on an aircraft of a landing gear retractable in flight permitted increasing the maximum speed by 10-15 km/h; however, the increase in weight of the construction induced by the necessity of installation of additional units for retracting and lowering the landing gear would make them unprofitable. The height of the landing gear is selected with such calculation in order to ensure a tail clearance angle of  $11^\circ 50'$  to the aircraft. With such a tail clearance angle the ends of blades of the propeller with the engine running are considerably far from the surface of the earth, which protects them from damages when landing on a cover with high vegetation or sinking the wheels (skis) in swampy ground (wavy snow). The tail clearance angle of the aircraft jointly with the setting angle of the wing provides creation for the aircraft without a rise of the tail on a takeoff run of an angle of attack of steep slope of  $14^\circ 10'$ , an angle at which the aircraft has good takeoff characteristics.

The tail wheel is self-oriented. With deflection at an angle of less than  $90^\circ$  it returns to a neutral position, which facilitates

the rectilinearity of taxiing and takeoff of the aircraft, and after breakaway there is eliminated vibration of the installation of the tail wheel, which causes vibration of the fuselage.

#### 4. Aerodynamic Characteristics of an Aircraft

Aerodynamic properties of an aircraft are represented by polars. The polars examined below correspond to conditions of lowering of the aircraft neglecting the effect of airflow.

##### Polar of an Aircraft with Pressed Slats and Undeflected flaps

From Fig. 26 one can see:

- 1) the angle of attack of zero lift ( $\alpha_0$ ) is a minus  $1^\circ$ , and the coefficient of drag is equal to 0.027;
- 2) the optimum angle of attack ( $\alpha_{HB}$ ) is equal to  $6^\circ$ , and the maximum efficiency is 11.3;
- 3) the critical angle of attack ( $\alpha^*$ ) is equal to  $18^\circ$ , and the coefficient of lift reaches a maximum value and is equal to 1.23; inasmuch as the tail clearance angle of the aircraft is equal to  $11^\circ 50'$  and the setting angle of the upper wing is equal to  $3^\circ$ , the landing angle of attack of the upper wing will be about  $15^\circ$ , i.e., considerably less the critical;
- 4) upon achievement of an angle of attack of  $16^\circ$  the slats are automatically advanced.

##### Polar of an Aircraft with Retracted Slats and Undeflected Flaps

With the release of the slats operation of the wing occurs with new regularity. In Fig. 27 it is clear that the advancement of the slats caused:

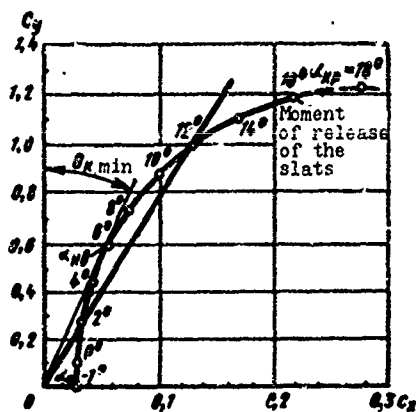


Fig. 26. Polar of an aircraft with extended slats and undeflected flaps.

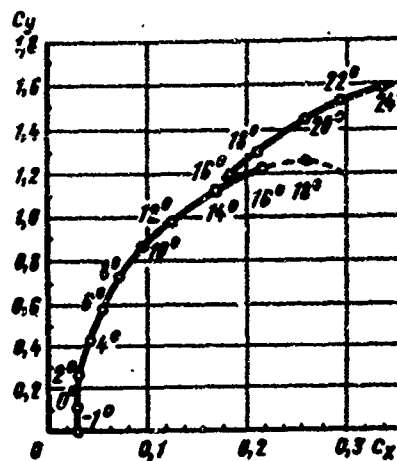


Fig. 27. Polar of an aircraft with retracted slats and undeflected flaps.

- 1) an increase in the critical angle of attack from  $18^\circ$  to  $24^\circ$ , and the maximum value of the coefficient of lift was increased from 1.23 to 1.67, which is 20%;
- 2) a decrease in the coefficient of drag at angles of attack from  $16^\circ$  to  $18^\circ$ , on the average of 18%;
- 3) an increase in the lift-drag ratio at angles of attack of  $16^\circ$ - $18^\circ$ , on the average of 30%.

#### Polar of an Aircraft with Extended Slats and Deflected Flaps of $40^\circ$

A change in airfoil camber with deflection of the flaps leads to a considerable change in the value of the characteristic points of the polar and aerodynamic coefficients. In connection with this Fig. 28 shows:

- 1) an angle of attack of zero lift decreases from minus  $1^\circ$  to minus  $11^\circ$ ;
- 2) a critical angle of attack decreases from  $18^\circ$  to  $14^\circ$ ;

3) the coefficient of lift ( $c_{y_{max}}$ ) is increased from 1.23 to 1.55, which is 26%;

4) the coefficient of drag in the whole range of the angle of attack is considerably increased;

5) the optimum angle of attack decreases to zero, and the maximum efficiency is 7.65, which is 32% less than the maximum efficiency with undeflected flaps.

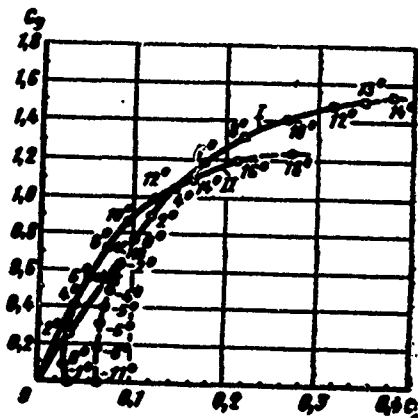


Fig. 28. Polar of aircraft with extended slats and deflected flaps of  $40^\circ$ :  
I - flaps are deflected;  
II - flaps are not deflected.

Polar of an Aircraft with Retracted Slats and Deflected Flaps of  $40^\circ$

With deflected flaps of  $40^\circ$  at an angle of attack equal to  $13^\circ$ , the slats are automatically advanced (Fig. 29), as a result of which the following occurs:

- 1) the critical angle of attack is increased from  $14^\circ$  to  $20^\circ$ ;
- 2) the maximum value of the coefficient of lift is increased from 1.55 to 1.95 and is 25%;
- 3) the coefficient of drag decreases after the advancement of slats: at an angle of attack of  $13^\circ$  it is equal to 0.285 and at an angle of attack of  $14^\circ$  - 0.298, which is 20% less than that with extended slats.

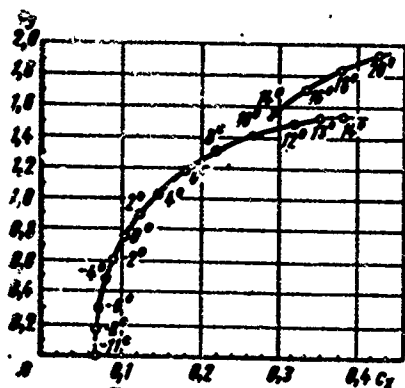


Fig. 29. Polar of an aircraft with retracted slats and deflected flaps.

The application of flaps and slats leads to an increase in range of the flying angles of attack, which amounts to:

- a) with undeflected flaps and advanced slats and deflected flaps and extended slats,  $25^\circ$ ;
- b) with deflected flaps and advanced slats,  $31^\circ$ .

The increase in the angular region of attack in the direction of large angles (less speeds) and, accordingly, the increase in coefficients of lift make it possible to produce descent of the aircraft with less forward velocity but with relatively great vertical velocity. A decrease in lift-drag ratio with deflected flaps leads to an increase in the angle and vertical velocity of descent. Thus, the separate and especially simultaneous operation of slats and flaps improves the landing characteristics of the aircraft.

Deflection of the flaps upon takeoff improves the takeoff characteristics of the aircraft. With deflection of the flaps on takeoff, because of the increase in the coefficient of lift necessary for breakaway of the aircraft, lift is created at a less speed, as a result of which the takeoff distance is reduced.

Release of the slats with exit of the aircraft at large angles of attack delays the separation of flow from the wing, which improves lateral stability of the aircraft.

## 5. Effect of Agricultural Equipment on Aerodynamic Characteristics of an Aircraft

The An-2 aircraft of the transport version possesses good aerodynamic properties; however with the installation on the aircraft of agricultural equipment the aerodynamic properties of it considerably worsen. The assembly of special equipment of the An-2 aircraft of the agricultural version includes a sprayer with an attachment for a separate supply of chemical poison and a duster with a tunnel or wing-tip duster.

With installation on the aircraft of agricultural equipment, the drag of it is increased.

Thus, for example, the coefficient of drag with testing in a wind tunnel of a duster with a tunnel duster was 0.0185. If one were to consider that with installation of it on an aircraft due to the interference the drag will be increased still by 25-30%, and that in the operation loading of the tank is possible with superphosphate or potassium salt of raised humidity, which causes the sticking of the chemical in the front section and directing channels of the duster, which causes an additional increase in drag, then the lift-drag ratio of the aircraft at all angles of attack will be considerably less as compared to that with a transport aircraft. Therefore, in comparing values of vertical velocities of aircraft of the transport and agricultural versions, other things being equal, there can be made the conclusion that the lift-drag ratio of aircraft of the agricultural version will be less than the lift-drag ratio of aircraft of the transport version on the average of 30%.

In confirmation of this Fig. 30 show the following:

1) the polar of the aircraft of the agricultural version with respect to the polar of the aircraft of the transport version shifted to the right by magnitude  $\Delta c_x$ , which taking into account the interference and effect of chemicals is about 0.02;



2) the optimum angle of attack is increased up to  $7^\circ$ , and the maximum efficiency of the aircraft decreases and will be approximately 8;

3) the value of the angle of attack of zero lift and critical angle is not changed.

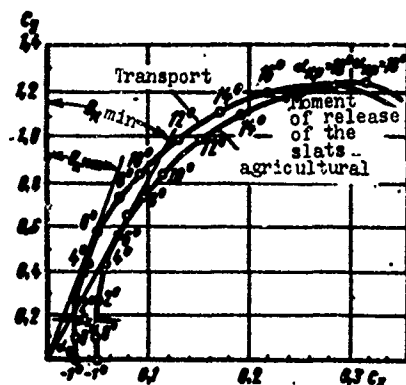


Fig. 30. Polar of aircraft of the agricultural version with a tunnel duster and taking into account the effect of humidity of chemical poisons.

## CHAPTER IV

### POWER SYSTEM

The power system is designed for the creation of a tractive force necessary for overcoming the drag of air and providing forward motion of the aircraft. The power system of the An-2 aircraft consists of an aircraft reciprocating engine of internal combustion ASh-62IR and a propeller. The engine converts thermal energy of the fuel into mechanical energy, expended for rotation of the propeller.

The propeller is a blade apparatus designed for the creation of tractive force.

Thrust of the propeller is reaction force. With rotation of the propeller its blades encounter air at a certain angle of attack, seize the masses of air and reject back, and acting on the blade is the reaction of rejected air, which is a tractive force.

#### 1. Altitude Performance and Basic Conditions of Operation of the Engine

The power at which parts of the engine are designed is called nominal, and it ensures obtaining the calculation maximum of speed of flight of the aircraft.

The prolonged continuous operation of the engine in normal rating causes thermal and mechanical overstress of its units in view of which the duration of operation of the engine in normal rating is limited in time and should not exceed one hour.

The presence on the engine of a centrifugal supercharger makes it possible to create on and in flight up to an altitude of 200 m a supercharging of 1050 mm Hg and to obtain a relatively high power. This power is called forced or takeoff and is used in exceptional cases when takeoff at normal rating of operation work of the engine does not ensure safety.

The duration of continuous operation of the engine in takeoff conditions should not exceed 5 minutes. With an increase in altitude of flight, starting from 200 m, the boost pressure ( $p_K$ ) drops decreasing by 10 mm with an ascent of each 100 m of altitude and taking into account the impact pressure at an altitude of 1670 m will be 900 mm Hg. However, because of the presence of a supercharger and decrease in counterpressure at the exhaust, the rated power of the engine with an increase in flight altitude up to the rated is increased.

The character of the change in rated power of the engine with respect to altitude is represented on a graph (Fig. 31), which is called altitude performance.

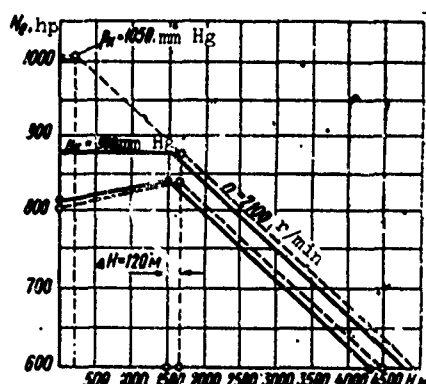


Fig. 31. Altitude performance of the engine ASh-62IR (solid line - neglecting the impact pressure; dashed line - taking into account the effect of impact pressure).

On altitude performance the following is shown:

- 1) on land the effective power of the engine ( $N_e$ ) is 820 hp;
- 2) at rated altitude (1500 m - neglecting the impact pressure, 1670 m - taking into account impact pressure) effective power of the engine is increased by 20 hp;

3) at altitudes higher than the rated the power of the engine decreases - at an altitude of 4500 m it will be 600 hp.

In all conditions less than operational operation of the engine is not limited in time.

Operational conditions are called such operating conditions of the engine at which the power of it is 0.9 of the nominal. Horizontal flight of the An-2 aircraft with indicated cruising speeds of 145-225 km/h is ensured by expenditure by the engine of 40 to 70% of the rated power. Operating conditions of the engine with the use of 70% of the rated power is called by the greatest cruising; with the use of 40% of the rated power - the least cruising.

Operation of the engine in basic conditions is characterized by parameters shown in Table 2.

Table 2.

Operating conditions of the engine	Power of the engine	Boost pressure, mm Hg	Revolutions per minute
Takeoff.....	1000	1050	2200
Nominal:			
at land.....	820	900	2100
at rated altitude.....	840	900	2100
Operational.....	738	830	2030
The greatest cruising power.....	615	760	1850
The least cruising power.	328	530	1500

## 2. Geometric and Kinematic Characteristics of the Propeller

The propeller consists of blades of wing-shaped form and boss. The blade of the propeller consists of fin and shank part. The blade, just as the wing, has a leading edge and trailing edge. The front more convex surface of the blade is called face, and the opposite surface - working side (Fig. 32a). The part of the blade, limited in length by two closely located sections is called by the element of the blade.

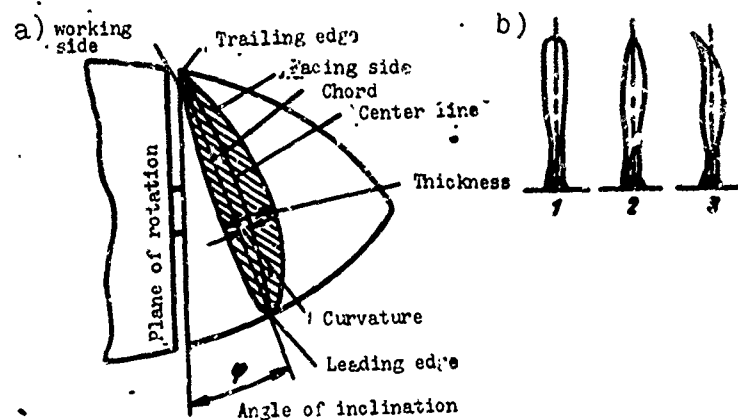


Fig. 32. Elements comprising the geometric characteristics of the blade: a) characteristic of profile of the blade; b) basic planforms of the blades: 1 - oar-shaped; 2 - symmetric; 3 - saber-shaped.

Operation of the blade depends on the blade planform, form of the profile, diameter, and angle of inclination of the blade. These elements and geometric pitch comprise the geometric characteristics of the blade.

According to the planform the blades are: saber-shaped, symmetric and rectangular (oar-shaped) - Fig. 32b. The profile of the blade of the propeller is characterized by the same elements as the profile of the wing: length of the chord, curvature and thickness. The greatest is seen in profiles which are thin biconvex and close to being symmetric.

The diameter of the propeller ( $D$ ) is called the diameter of a circle described by ends of blades of the propeller. Contemporary propellers have a diameter within 2-5 m. The plane perpendicular to the axis of rotation is the plane of rotation of the propeller. The angle of inclination of the blade of the propeller ( $\phi$ ) is the angle between the plane of rotation and direction of the chord of the element of the blade of the propeller. The blade has a geometric twist, and therefore angles of inclination with distance of elements of the blade of the propeller from the shank part decrease.

The angle of inclination of the blade of the propeller can be considered to be the angle of inclination of the element of the blade of the propeller located 1000 mm from the axis of rotation of the propeller. Propellers used on the An-2 aircraft have the following geometric characteristics (Table 3).

Table 3.

Element of the characteristic	Designation of propeller			
	V-514-D8	AV-2	V-509-D-9A	AV-7N-161
Number of blades.....	4	4	4	3
Form of the blade.....	Oar-shaped			Symmetric
Diameter of the propeller, m.....	3.6	3.6	3.6	3.6
Angle of inclination of the blade on the radius equal to 1000 mm, deg:				
minimum.....	16	17	13.5	19
maximum.....	31	32	28.5	44
in the position "Reverse".....	-7	-	-	-

Geometric propeller pitch (H) is called the distance passable by the propeller in one revolution in an inflexible medium:

$$H = 2\pi r \operatorname{tg} \varphi.$$

Inasmuch as the propeller revolves in an inflexible medium, which is air, then in one revolution it passes less distance. The distance passable by the propeller in one revolution in air is called the advance ratio of the propeller:

$$A = \frac{V}{n},$$

where V – flight speed of the aircraft, m/s; n – number of revolutions of the propeller per second.

The difference between the geometric pitch and advance ratio of the blade is called slip:

$$S = H - A.$$

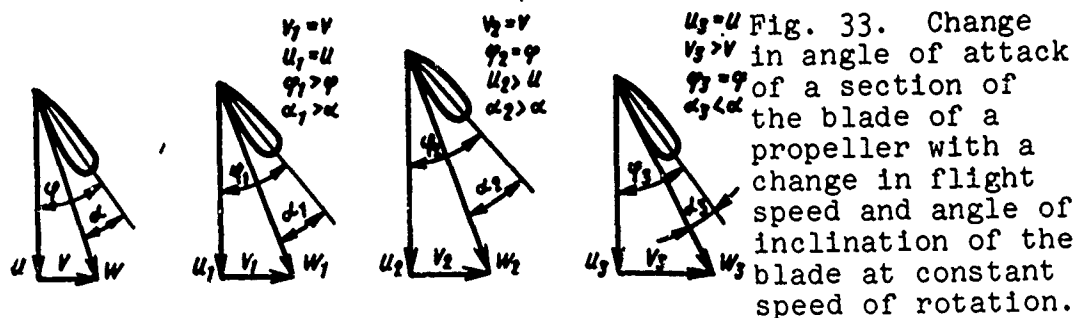
With an increase in flight speed of the aircraft slip of the propeller decreases, and with flight of the aircraft at high speeds (diving) it can be negative. With positive slip the propeller, seizing and rejecting the air back, creates a positive thrust, and with negative slip, rejecting the air forward - drag. When the slip is equal to zero, and the profile of the blade is symmetric, the blade will not create thrust.

The advance ratio and slip characterize the ability of the propeller to create a tractive force and therefore comprise kinematic characteristics of the propeller.

### 3. Aerodynamic Forces of a Propeller

Having a wing profile, the blade of a propeller during interaction with the airflow, according to the same laws of aerodynamics as those of the wing, creates an aerodynamic force. The magnitude and direction of the aerodynamic force depend on the angle of attack of the blade of the propeller. Inasmuch as the blade of the propeller in contrast to the wing is simultaneously under the effect of forward velocity ( $V$ ) and peripheral velocity ( $u$ ), then the angle of attack of the element of the blade of the propeller is considered the angle between the direction of the chord of the element and resultant speed ( $W$ ).

The angle of attack of the element of the blade of the propeller depends on the forward and peripheral velocity of the angle of inclination of the blade. Figure 33 shows that with an increase in peripheral velocity ( $u$ ) and angle of inclination ( $\phi$ ) the angle of attack is increased; with an increase in forward velocity ( $V$ ) the angle of attack ( $\alpha$ ) decreases.



With the flowing around of the element of the propeller blade by airflow at speed ( $W$ ) with an angle of attack ( $\alpha$ ), because of the difference in pressures between the leading edge and trailing edge of the flowing around of the working and face side and also friction of air in the boundary layer on the element of blade, there appears total aerodynamic force ( $R_{эЛ}$ ).

Figure 34a shows that if one were to decompose the total aerodynamic force of the element of the propeller blade into two directions, then there can be obtained:

- a) in a direction parallel to the axis of rotation of the propeller — an elementary tractive force ( $\Delta P$ );
- b) in a direction opposite the rotation of the propeller — resisting force to rotation of the element of the blade ( $\Delta Q_{сп.эЛ}$ ).

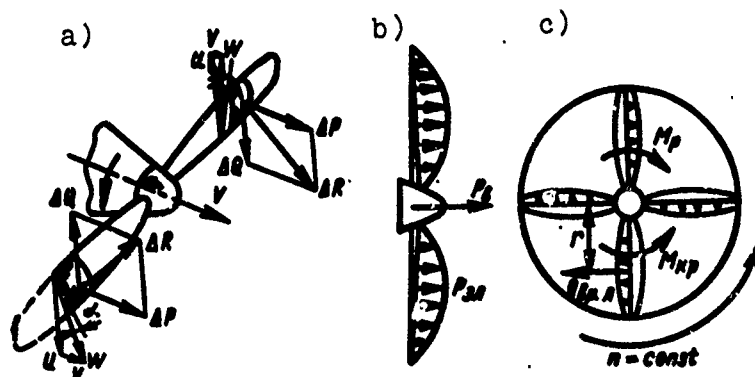


Fig. 34. Aerodynamic forces of the element of a propeller blade and propeller as a whole: a) aerodynamic forces of the element of the propeller; b) tractive force of the propeller; c) resisting force to rotation of the propeller.

The sum of the elementary tractive forces applied to the axis of rotation of the propeller, comprises the tractive force of the propeller (Fig. 34b) and is determined by the formula

$$P_{э} = a_p n^3 D^4,$$



where  $\alpha$  - thrust coefficient considering the planform of the blade form of the profile, degree of treatment of the blade and angle of attack;  $\rho$  - air density;  $n$  - revolut of per second;  $D$  - diameter of the propeller, m.

The sum of the elementary resisting forces to rotation consists of the resisting force to rotation of the propeller blade. Inasmuch as the blades have geometric symmetry, the magnitude and removal of resisting forces to rotation of the blades will be identical, which is shown in Fig. 34c. The magnitude of the resisting force to rotation of the blade of the propeller is determined by the formula

$$Q_{\text{p.1}} = c_{x_{\text{II}}} \frac{\rho W^2}{2} S_{\text{II}},$$

where  $c_{x_{\text{II}}}$  - coefficient considering the form of the blade and form of the profile, degree of treatment of the blade and angle of attack;  $W$  - resultant speed;  $S_{\text{II}}$  - area of the blade.

The resisting force to rotation of the whole screw is determined by the product of the resisting force to rotation of the blade by the number of blades (1):

$$Q_{\text{p.0}} = Q_{\text{p.1}} i.$$

The resisting force to rotation of the propeller with respect to the axis of rotation of the propeller creates an antitorque moment of the propeller (reactive moment), which with steady rotation of the propeller ( $n = \text{const}$ ) is balanced by torque of the engine (Fig. 34c).

$$M_p = Q_{\text{p.0}} r; \quad M = M_{\text{xp}}; \quad M_{\text{xp}} = 716,2 \frac{N_e}{n},$$

where  $M_p$  - reactive moment of the propeller;  $M_{\text{xp}}$  - torque of the engine  $N$  - power of the propeller  $n$  - number of revolutions of the propeller.

#### 4. The Power Necessary for Rotation of the Propeller and Its Thrust Housepower

The power necessary for overcoming resisting forces to rotation of the propeller per unit of time is called the power necessary for rotation of the propeller:

$$N_{\text{r.}} = i \frac{Q_{\text{r}} u}{75},$$

where  $Q_{\text{r}}$  - resisting force to rotation of the blade, hp;  $u$  - peripheral velocity, m/s;  $i$  - number of blades of the propeller.

The resisting force to rotation of the given propeller depends on the speed of flight, revolutions and altitude of flight. With an increase in the speed of flight and altitude, the resisting force to rotation and, consequently, the power necessary for rotation of the propeller decrease. With an increase in the number of revolutions because of the increase in angle of attack the resisting force to rotation of the propeller and the power necessary for rotation of the propeller are increased.

The propeller, absorbing the energy transmitted to it by the engine simultaneously develops a thrust, which overcomes the resisting force of the aircraft. The work produced by the tractive force of the propeller in the process of forward motion of the aircraft in one second is called traction or net power of the propeller.

Traction power of the propeller is determined by the formula

$$N_{\text{r.}} = \frac{P_{\text{r}} V}{75}.$$

With conservation of the constancy of revolutions thrust of the given propeller depends on the altitude and speed of flight. With an increase in altitude of flight (because of a decrease  $\rho$ ) and with an increase in speed of flight (because of a decrease in  $\alpha$ ) the thrust of the propeller decreases. With flight of the aircraft at a speed close to double the maximum speed, the thrust of the propeller is equal to zero. With operation in place the propeller

develops maximum thrust (AV-2 - 1850 kgf, V-509-D-9A - 1600 kgf), but inasmuch as the speed is equal to zero the thrust horse power of the propeller is equal to zero. With flight of the aircraft at a speed close to double the maximum, the thrust of the propeller is equal to zero, and therefore thrust horse power of the propeller will also be equal to zero. Thus the thrust horsepower of the propeller achieves a maximum value during flight of the aircraft at maximum speed, since the combination of maximum speed of flight with nominal revolutions creates on blades of the propeller the optimum angle of attack.

## 5. Efficiency of the Propeller

The efficiency of the propeller is called the number showing what part of the power of the engine expended for rotation of the propeller is turned into thrust horsepower of the propeller is designated by the letter  $\eta$  and is determined by the formula

$$\eta = \frac{N_{r.p.}}{N_e},$$

where  $\eta$  - efficiency of the propeller;  $N_{r.p.}$  - thrust horsepower of the propeller, hp;  $N_e$  - effective power of the engine, hp.

Inasmuch as the maximum value of thrust horsepower of the propeller is attained during flight of aircraft at maximum speed, then the efficiency will also be maximum.

The magnitude of efficiency of the propeller depends on revolutions of the propeller and speed of flight, since the change in them involves a change in the angle of attack on blades of the propeller. At constant nominal revolutions with an increase in flight speed the angle of attack on blades of the propeller decreases, approaching the most optimum, as a result of which the efficiency of the propeller is increased, which is shown in Fig. 35.

From the figure it is clear that the efficiency of the propellers used on the An-2 aircraft with an increase in flight speed up to the

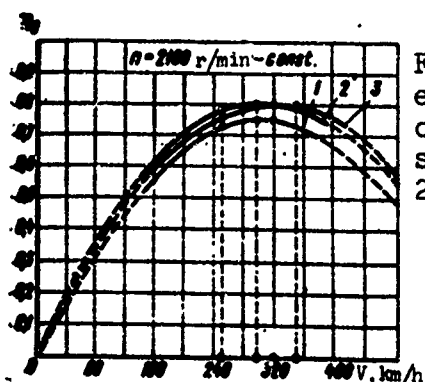


Fig. 35. Dependence of the efficiency of propellers used on the aircraft on flight speed: 1 - V-509-D-9A; 2 - AV-2; 3 - AV-7N-161.

rated (AV-7N-161 up to 320-350 km/h, with AV-2 and V-509-D-9A up to 300-320 km/h) is increased, and at calculated speeds at which on blades of the propeller there is created the optimum angle of attack it reaches a maximum value: AV-2 and AV-7N-161 - 0.8, and V-509-D-9A - 0.77.

With a further increase in the flight speed the angle of attack on blades of the propeller becomes less optimum and the efficiency of the propeller decreases.

With flight of the aircraft at maximum speed, which at the rated altitude is equal to 254 km/h, the efficiency of the propeller will be: AV2 - 0.77, AV-7N-161 - 0.75, V-509-D-9A - 0.73. This means that the propellers used on the An-2 aircraft turn on the average of only 75% of the power of the engine into thrust horsepower of the propeller, and 25% of the power of the engine is expended to overcome the profile, induced and wave drags. The magnitude of the profile drag depends on the form and thickness of the profile, degree of treatment of the surface of the propeller blade. Less drag is created by thin symmetric profiles. The magnitude of induced drag ( $Q_i$ ) depends on the blade planform: less  $Q_i$  is created by a blade of saber-shaped form and greater - by rectangular (oar-shaped) form.

The magnitude of wave drag depends on the speed of flight, number of revolutions and form of the blades. With an increase in speed of flight and number of revolutions higher than the calculated, the wave drag is increased. To decrease the peripheral velocity of

the propeller at constant engine revolutions, reduction gears are used. The presence of a reduction gear in the engine ASh-62IR with a reduction ratio of 11/16 and a comparatively small diameter of the propellers ensure during operation of the engine in normal rating the creation on ends of the blades of a resultant speed considerably less than the speed of sound, which improves condition of operation of the propeller.

## 6. Available Power of the Propeller

The available power of a propeller is called the power developed by the propeller in normal rating of operation of the engine, and it is determined by the formula

$$N_p = \eta N_e.$$

From an analysis of the formula the following conclusions can be made:

1) with an increase in flight speed (because of an increase in efficiency) and altitude of flight up to the rated (because of an increase of effective horsepower of the engine) the available power of the propeller is increased;

2) with flight of an aircraft at an altitude higher than the rated and at a speed greater than the maximum, the available power decreases.

The character of the change in available power of the propeller depending upon speed and altitude of the flight is shown in Fig. 36.

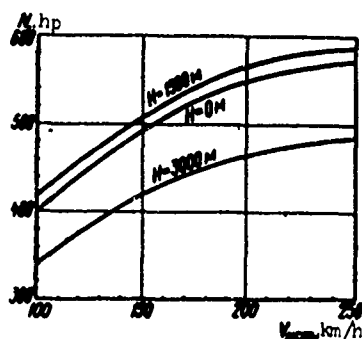


Fig. 36. Dependence of available power of the power system on forward velocity and altitude of flight of the aircraft.

So that the propeller develops high power, at a definite position of the engine control lever it is necessary with respect to the speed of flight of the aircraft (supercharging) to select and set the revolutions so that the combination of them with the speed of flight creates an angle of attack on blades of the propeller close to the optimum (Fig. 37).

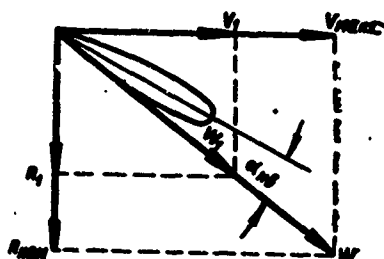


Fig. 37. Combination of optimum revolutions with flight speed (supercharging) when the angle of attack on blades of the propeller is close to the optimum.

It is shown on the figure that the speed of flight ( $V_1$ ) corresponds to the optimum revolutions ( $n_1$ ).

#### 7. Principle of Operation of a Propeller of Variable Pitch and the Advantages of It Over a Fixed-Pitch Propeller

For understanding the necessity of application of propellers of variable pitch (VISH) and the principle of operation of it, it is expedient in the beginning to examine the operation of a propeller of fixed pitch (VFSh). A propeller of fixed pitch is a propeller the blades of which can change the angle of inclination only on land, and the angle of inclination of the blades is set with such calculation that with the flight of an aircraft at the calculated speed the efficiency of it is maximum.

For the characteristic of operation of a propeller of fixed pitch graph (Fig. 38) are given.

On the graph — Fig. 38a — the following is shown:

- 1) with an increase in flight speed the effective power of the engine ( $N_e$ ) is somewhat increased owing to the increase in high-speed

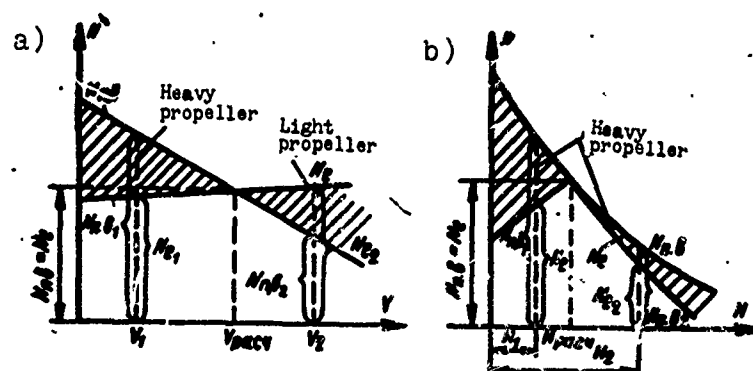


Fig. 38. Graphs of the operation of propeller of fixed pitch: a) joint operation of the propeller and engine at different flight speeds; b) joint operation of the propeller and engine at different flight altitudes.

supercharging, and the required power for rotation of the propeller ( $N_{p.p.}$ ) owing to the decrease in angle of attack ( $\alpha$ ) on blades of the propeller decreases;

2) at speeds of flight less than the calculated, when the angle of attack on blades of the propeller becomes greater than the optimum, the power necessary for rotation of the propeller is greater than the effective power of the engine, as a result of which the propeller becomes "heavy";

3) at flight speeds greater than the calculated, when the angle of attack on blades of the propeller becomes less than the optimum, the power necessary for rotation of the propeller is less than the effective, and the propeller becomes "light"; in order not to allow acceleration (increase in revolutions higher than the maximum permissible, which can involve an accident of the engine) the pilot is forced to decrease the power of the engine.

In the graph — Fig. 38b — the following is shown:

1) with ascent up to the rated altitude the effective power of the engine is increased, and the required power for propeller rotation decreases;

2) at the rated altitude the required power for propeller rotation is equal to the effective power of the engine;

3) at altitudes greater than the rated the effective power of the engine decreases by a larger magnitude than the required power for rotation of the propeller.

Thus the propeller at all altitudes of flight besides the rated is "heavy."

From an analysis of the graphs the conclusion can be made that the propeller of fixed pitch does not permit completely using the power of the engine in all range of speeds and altitudes of flight of the aircraft besides the rated, as a result of which flying data of the aircraft worsen.

Deficiencies of the propeller of fixed pitch are eliminated by application of the propeller of variable pitch. The principle of operation of the propeller of variable pitch consists in the fact that it, being connected with the constant-speed control unit (RPO), automatically changes the angle of inclination of blades of the propeller in flight, as a result the angle of attack and, consequently, the power necessary for rotation of the propeller are changed. The change in the angle of inclination of the blades is produced by such a magnitude that with a new value of the resultant speed and an angle of attack, the power necessary for rotation of the propeller is equal to the effective power of the engine at the given speed and altitude of flight.

From formula  $H = 2\pi r \operatorname{tg} \phi$  it is clear that with an increase in the angle of inclination of blades ( $\phi$ ) the propeller pitch is increased. Therefore, it is accepted to call the turn of the blades for an increase in the angle of inclination - the setting of the propeller at "high pitch" and the turn of blades for the decrease in the angle of inclination - the setting of the propeller at "low pitch."



According to the principle of the turn of blades of the propeller of variable pitch, a direct, reverse and double operating scheme occur. For a propeller of a direct operating scheme the turn of the blades at "low pitch" is carried out under the pressure of the oil and at "high pitch" – under the action of the moment of centrifugal forces created by counterpoises. For a propeller of a reverse scheme of action the turn of the blades at "low pitch" is produced under the action of the moment of centrifugal forces created by blades of the propeller and at "high pitch" – under the pressure of the oil. For the propeller of a double scheme of action the transition of blades to "high and low pitch" is produced under the pressure of the oil and centrifugal forces.

Figure 39a shows that every point on the blade of the propeller with its rotation tries to be detached under the action of its inertial forces.

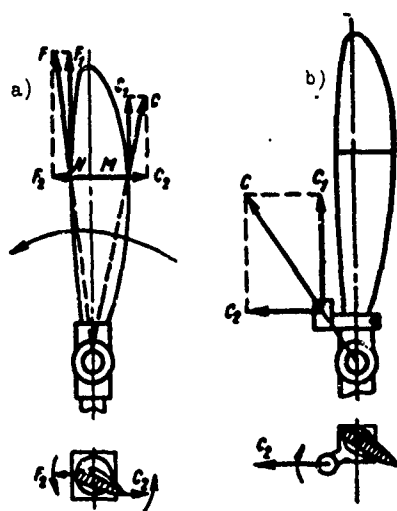


Fig. 39. Operation of propeller of variable pitch: a) action of inertial forces of the blade of the propeller; b) action of centrifugal forces of counterpoise for a propeller operating along a direct scheme.

Components of inertial forces  $F_1$  and  $C_1$  try to pull the blade from the hub of the propeller and components  $F_2$  and  $C_2$  – set the blade in the plane of rotation of the propeller – transfer it to "low pitch."

Figure 39b shows that with the setting of the counterpoise at a definite angle the moment of its centrifugal forces overcomes

moments of centrifugal forces of the blade and turns the blade to "high pitch." The propeller of variable pitch, operating according to a reverse scheme, is not used in practice, since the case of failure of the oil system acceleration of the propeller occurs. Variable-pitch propellers AV-2, and AV-7N-161 operate on the scheme of double action and the propeller V-509-D-9A — on a direct scheme.

Application of the propeller of variable pitch considerably increases the flying characteristic of the aircraft as compared to the propeller of fixed pitch. The improvement of flying data aircraft with the application of a propeller of variable pitch is expressed by the following:

- a) decrease in takeoff distance and increase in vertical velocity of ascent by 30-40%;
- b) increase in ceiling and payload by 15%;
- c) increase in distance and duration of flight by 20%.

#### 8. Operation of a Propeller of Variable Pitch and Its Control in Different Flight Conditions

So that the propeller is not heavy and the engine could develop revolutions corresponding to takeoff conditions, before takeoff the pilot sets the propeller at "low pitch." With an increase in speed of the aircraft in the process of a takeoff run the turns will be maintained, since with a decrease in angles of attack on blades of the propeller the constant-speed control unit will shift the blades at large angles of inclination, increasing the angles of attack, and consequently, the required power for rotating the propeller up to an equivalence with the effective power of the engine will be increased. The angle of attack on blades of the propeller owing to the increase in resultant speed is somewhat less than the initial, which is shown in Fig. 40.

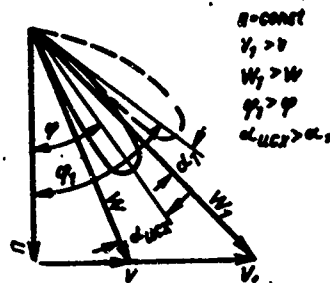


Fig. 40. Change in magnitude of the angle of attack of the blade of the propeller of variable pitch with a change in flight speed.

To ensure maximum vertical rate of ascent when climbing the pilot sets the optimum speed and then for obtaining the greatest power of the propeller at a given speed sets the optimum revolutions. With ascent to altitude up to the rated the power of the engine is increased, and the required power for rotation of the propeller decreases; however, the revolutions are maintained, since the constant-speed control unit changes the blades to large angles of inclination. After the rated altitude, inasmuch as the power of the engine decreases by a greater magnitude than the power necessary for rotation of the propeller, the constant-speed control unit changes the blades of the propeller from "high" to "low pitch."

In horizontal flight, in spite of oscillations in speed, the revolutions are maintained constant since the constant-speed control unit continuously changes the angle of slope of blades in a certain direction. Upon achievement of the rated altitude and maximum speed of flight of the aircraft, the blades of the propeller are set on rests of "high pitch." With an increase in speed or altitude of flight at which the range of turn of the blades is calculated, the propeller of variable pitch will operate as a propeller of fixed pitch. So that the engine can develop revolutions corresponding to takeoff or normal rating in the case of balked landing, the pilot after a fourth turn prior to the span of obstacles, but at an altitude of not less than 50 m, changes the propeller to "low pitch."

To avoid detonation with control of the engine and propeller it is necessary:

- a) with an increase in power of the engine — to set the

revolutions and then the supercharging;

b) with a decrease in power of the engine -- to decrease the supercharging and then set the revolutions.

### 9. Drag of the Propeller with Engine Failure

With failure of the engine in flight the drag of the propeller is increased, the pilot passes to the optimum rate of descent, and the revolutions of the propeller, if it autorotates, sharply decrease, which leads to the appearance of negative angles of attack on blades of the propeller. Figure 41 shows that force  $Q$  acts in the opposite direction of the motion of the aircraft, and force  $A_p$  is the force under the action of which the propeller autorotates.

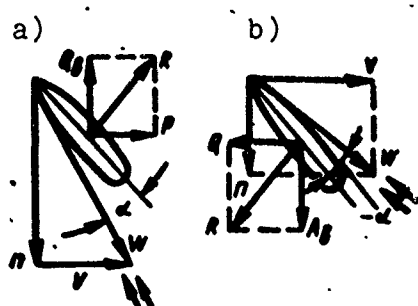


Fig. 41. Causes of the increase in drag of the propeller with failure of the engine: a) direction of aerodynamic forces in normal flight; b) direction of aerodynamic forces with engine failure.

Drag of the propeller of a nonoperating engine considerably lowers the lift-drag ratio of the aircraft, which one should consider in the determination of the distance of gliding when a site for forced landing is selected. The magnitude of drag of a propeller of a nonoperating engine depends on the angle of inclination of the blades, speed of flight and on the weather, the propeller autorotates or stops.

With descent of the aircraft at average speeds the drag of an autorotating propeller is less than one stopped and at low speeds (during landing) -- more.

The drag of a propeller stopped at "low pitch" is greater than that for a propeller stopped at "high pitch." Tests showed that the drag of a propeller stopped at "high pitch" is 40-50% of the drag of a propeller stopped at "low pitch."

Thus, for the achievement of great distance of gliding in the case of engine failure in flight it is necessary that the blades be set at "high pitch."

In the determination of the distance of gliding at the optimum speed, the efficiency of the aircraft should be taken at not more than 9.

#### 10. Special Propellers of Variable Pitch and Their Application

Special propellers used on the An-2 aircraft are the feathering propeller AV-7N-161 and reversible-pitch propeller V-514-D8. The feathering propeller is a propeller of variable pitch, the blades of which can be set on the flow, which decreases the drag of it in the case of failure of the engine. The drag of a feathered propeller is 5-10% of the drag of a propeller stopped at "low pitch." The feathering position of the propeller AV-7N-161 is  $90^{\circ}30'$ ; on the An-2 aircraft it is not functional. The reversible-pitch propeller is a propeller of variable pitch capable of creating a negative thrust, since the blades of it can be changed to negative angles of inclination (Fig. 42). The reversible-pitch propeller V-514-D8 is used on the An-2V aircraft for decreasing the landing run of the aircraft during landing on a water surface and improving the maneuverability with motion on the water. With a change in the propeller to the position "Reverse" and short-term giving of the gas up to the full, the landing run of the aircraft is reduced by 35-40%.

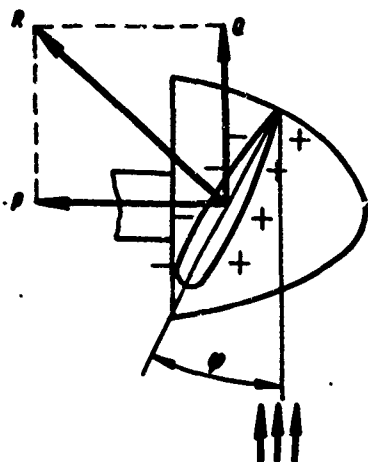


Fig. 42. Principle operation of a reversible-pitch propeller.

# 11. Effect of Operation of the Engine on Aerodynamic Characteristics of an Aircraft

The engine has a considerable effect on the improvement of the aerodynamic properties of an aircraft. This effect is greater, the larger the angle of the wing setting and operating conditions of the engine. At angles of attack larger than the optimum the axis of the propeller does not coincide with the direction of flight and creates the condition of oblique airflow which causes a change in magnitude and direction of the full aerodynamic force on blades of the propeller. In Fig. 43 it is clear that for the lowering blade the resultant speed ( $W_{on}$ ) and angle of attack ( $\alpha_{on}$ ) are greater than they are for the raising blade. The aerodynamic force of the lowering blade ( $R_{on}$ ) is greater than that for a raising ( $R_{noa}$ ), and the resultant aerodynamic forces of blades ( $P_{act}$ ) — tractive force — is deflected from the axis of rotation of the propeller upwards on angle  $\beta$ .

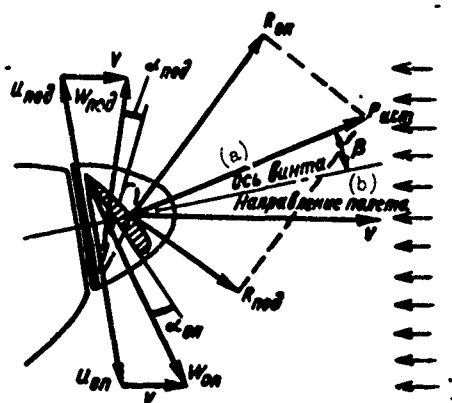


Fig. 43. Diagram of the operation of a propeller with oblique airflow.

KEY: (a) Axis of the propeller; (b) Direction of flight.

Figure 44 shows that if one were to decompose the tractive force into components in the direction of flight and perpendicular to it, the component  $P_Q$  is expended for overcoming drag of the aircraft, and component  $P_y$ , coinciding with the direction of action of lift of the wing, increases it.



Fig. 44. Effect of oblique airflow of the propeller on the magnitude of lift of the wing.

KEY: (a) Axis of the propeller

Furthermore, in the propeller's operation aerodynamic forces of the aircraft are increased because of the increase in local velocity of the flowing around of parts of the aircraft on velocity of stream rejected by the propeller back. An increase in local velocity does not have essential effect on the efficiency of the aircraft, since lift and drag are changed in proportion to the square of the speed. Consequently, the aerodynamic properties of the aircraft with operation of the propeller are improved because of the increase in lift of the wing by the magnitude of vertical component of thrust  $P_y$ .

On polars of the An-2 aircraft, taking into account the effect of the propeller operation represented on Fig. 45, one can see the following:

1) the larger the angles of attack, the greater the increase in coefficient of lift ( $c_y$ ): at an angle of attack equal to  $18^\circ$ ,  $c_y$  in gliding is 1.23, and in takeoff operating conditions of the engine - 1.7, whereas at an angle of attack equal to  $10^\circ$ ,  $c_y$  in gliding is equal to 0.88, and in takeoff operating conditions of the engine - 1.2. In the first case the difference in value of  $c_y$  is 0.47 and in the second case - 0.32;

2) the higher the operating conditions of the engine, the greater the effect of the vertical component of thrust on the increase in  $c_y$ : at an angle of attack equal to  $10^\circ$ , the increase in

$c_y$  in takeoff operating conditions of the engine is 0.32, and in normal rating - 0.29;

3) the increase in  $c_y$  owing to the increase in local velocity of the flowing around of parts of the aircraft occurs considerably less than that of  $c_v$ : at an angle of attack equal to  $18^\circ$ , in normal rating of operation of the engine  $c_x$  is 0.029 and in takeoff operating conditions of the engine - 0.031.

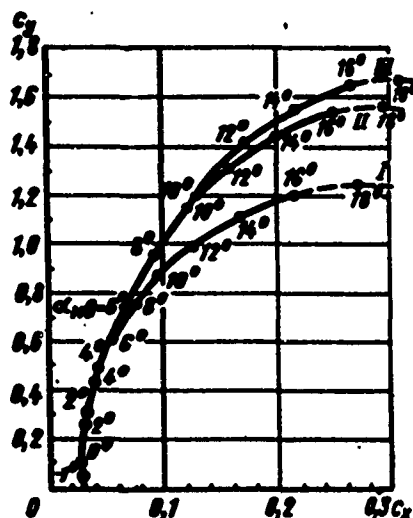


Fig. 45. Polars of the aircraft: I - neglecting propeller operation; II - taking into account the operation of the propeller in normal rating; III - taking into account the operation of the propeller in takeoff conditions.

Because of the improvement of aerodynamic properties of aircraft with operation of the power system the following decrease:

- a) lift-off speed by 17%;
- b) takeoff distance by 25-30%;
- c) takeoff distance and necessary speeds of flight at characteristic angles of attack, since the lift because of the propeller operation is increased by 10-15%.



## CHAPTER V

### BALANCE, STABILITY AND CONTROLLABILITY OF AIRCRAFT

#### 1. Center of Gravity and Position of Center of Gravity of Aircraft

The point of application of resultant force of weight of all aircraft parts is called the center of gravity (CG), position of CG on the aircraft is usually determined by the method of double weighing. Aircraft is successively installed on scales in two positions, as shown on Fig. 46; at each weighing there are measured the readings of the front and rear scales. By knowing the distance between scales and the forces affecting the front and rear scales in both cases, by rules of mechanics we determine the amount of resultant force and the line of its action for each of these positions of the aircraft. The point of intersection of lines of action of resultant  $A$  and  $A_1$  is the center of gravity of the aircraft.

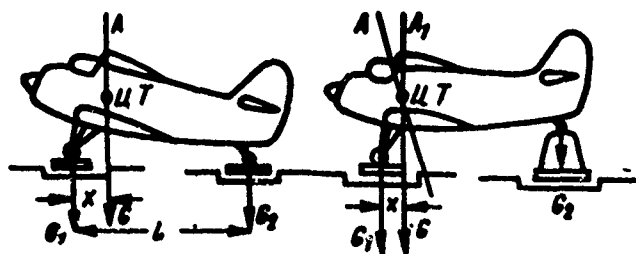


Fig. 46. Determination of position of CG of aircraft by weighing.

The position of center of gravity on the aircraft is oriented

relative to the mean aerodynamic chord of the wing. The mean aerodynamic chord of the wing (MAC) is the chord of a rectangular wing, which has area, amount of total aerodynamic force and position of CG that are identical to the given wing at equal angles of attack.

In Fig. 47 there are shown MAC coordinates of An-2 aircraft: the leading edge is located 660 mm higher than the horizontal datum line of the aircraft and 50 mm behind the fifth frame of the fuselage, the trailing edge - 586 mm higher than the horizontal datum line. Length of MAC is 2.27 m.

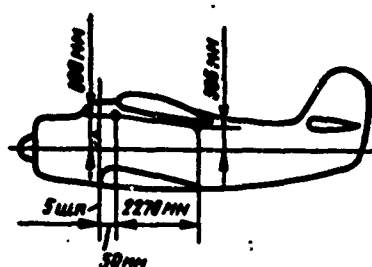


Fig. 47. MAC coordinates of cell of aircraft wing.  
[Translators note:  $\text{mm}$  = fr = frame].

The position of center of gravity relative to the leading edge of MAC, expressed in percent of its length (b), is called the aircraft CG position ( $\bar{x}$ ).

$$\bar{x} = \frac{x}{b} \cdot 100,$$

where  $x$  - distance from the leading edge of MAC to the center of gravity, m;  $b$  - length of MAC, m.

With the exception of aircraft equipped with 12 passenger seats, arranged according to the flight (depending upon the series), the CG position of an empty An-2 aircraft with total equipment is 20.4-22.4% of MAC. The position of aircraft CG is changed with servicing of the aircraft with fuel and oil, and arrangement of crew, passengers and cargo in the aircraft.

Since rotation of the aircraft occurs relative to the center of gravity, then the position of it on the aircraft will essentially affect the behavior of the aircraft in air and its control.

To ensure flight safety on An-2 aircraft of the land version the following CG limits are established:

a) on aircraft up to the 60 series: maximum forward - 19.2% MAC; maximum aft - 32.2% MAC;

b) on aircraft higher than 60 series: maximum forward - 17.2% MAC; maximum aft - 33% MAC.

The range of operational CG limits on An-2 aircraft higher than 60 series is expanded due to increase of the area of horizontal tail surfaces and increase of the elevator angle upwards.

In industrial subdivisions of civil aviation there are operated An-2 aircraft up to the 60 series, on which at repair air bases there are installed stabilizers of increased area and the elevator angle is increased upwards. Maximum CG limits of such aircraft correspond to CG limits of aircraft above the 60 series, which is indicated in for forms.

The recommended CG range, at which the aircraft possesses the best flight characteristics is 23-28% of MAC.

## 2. Distribution of Cargo in Aircraft and Calculation of Its CG Position

Incorrect distribution of cargo in an aircraft leads to impairment of aircraft stability and controllability, lowers safety during takeoff, landing and balked landing, and a loose load during takeoff or in flight can be shifted back to frame No 15 and lead to severe disturbance of the CG position of the aircraft, loss of longitudinal controllability and stall of the aircraft. Therefore, before flight it is very important to correctly place the load, to secure it and to determine the position of CG of the loaded aircraft.

When loading the aircraft it is necessary to follow requirements of the order of the chief of Main Administration for Civil Aviation

(GUGVF) No 192 from 3 April 1961 and order of the Minister of Civil Aviation of USSR No 525 from 4 August 1965, according to which:

1. Gross weight for all versions of the aircraft should not exceed 5250 kg, payload - 1500 kg in the cargo version or 12 passengers in the passenger version.

2. Distribute cargo in the cargo version of the aircraft according to marks placed on the right side of the cargo section of the fuselage that are green and red in color (Fig. 48): opposite the green pointers with the inscription "Up to 1500 kg" place any load weighing up to 1500 kg, in this case the position of aircraft CG (at matching of center of the load with the pointer) will be 24-25% of MAC.

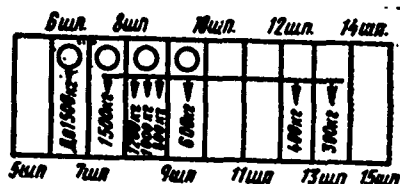


Fig. 48. Marks placed on the right side inside the cargo compartment of the aircraft from the 60 series, indicating maximum permissible aft arrangement of cargo  
[До = Up to; мм = fr = frame; кг = kg]

3. Place loads with large dimensions any place between the green pointer with mark "Up to 1500 kg" and read pointers with marks: 1500, 1200, 1000, 800, 600, 400 and 300 kg; the center of gravity of the load should be in front of the pointers, and the weight of the load should not exceed that designat under the given pointer.

If a load weighing 400, 600, 800 kg etc. is placed in the cargo compartment opposite the corresponding figures with a red pointer is such a manner that the center of gravity of the load is opposite the pointers, then such distribution would create maximum permissible

CG limit of the aircraft. Accomodation of a load weighing more than is shown under the pointer will create a CG exceeding the established limits.

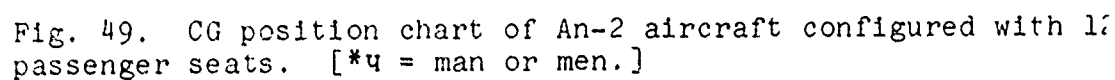
4. It is prohibited to place loads in the tail section of the fuselage, since arrangement of a load weighing 80 kg or location of one passenger at frame No 15 increases the CG by 2.5% of MAC.

5. With an incomplete amount of passengers, leave the rear seats vacant (set passengers with children in all cases on the front seats), and place cargo and baggage between frames No 6 and 8. On aircraft equipped with 12 passenger seats, arranged according to the flight, place the load closer to frame No 5.

6. After distribution of cargo assure its reliable attachment by cables to flc      ings and brackets in the sides of the hold; brief passengers so that during takeoff and landing they use seat belts and do not move through the cargo compartment during flight.

Before departure the CG position of the loaded aircraft is checked by the CG position chart. The CG position is computed on the CG position chart by the method of graphic addition of static moments, created by the types of load of the aircraft.

In Fig. 49, in the upper part of the chart, there is plotted a scale of CG positions and weight of the empty aircraft (a), to the right of it - diagram of layout of seats in the passenger compartment (b); in the upper right corner is placed the table - weight and CG position of empty aircraft (c). In this table the pilot is obligated to enter the weight and CG position of empty aircraft on which he flies, having refined them on the form. In the central part of the chart are placed scales for types of loading, shown vertically (d); each scale has a specific scale value, shown for pointers (e), showing the direction in which the reading should be taken. For a more precise reading the scale value is divided into intermediate graduations: on the left is shown small scale value; on the right - large.



With full load of passengers there is provided a total scale - "Seats 1-12." If the load center of gravity is arranged between two frames, then during the reading it is necessary to take the average scale value between these frames.

Lower part of the chart (f) shows the final result - CG position in % of MAC, depending upon the gross weight of the aircraft. The shaded part of the chart indicated CG positions exceeding the permissible limits.

Calculation of CG position is performed in the following way.

From point 1, located at the intersection of the line of center of gravity of the empty aircraft with the line of weight of the empty aircraft (upper scale), a vertical is lowered to the corresponding horizontal loading scale (2), and then is shifted along the horizontal scale in the direction shown by the pointer to the graduation corresponding to loading. From this graduation a vertical is lowered to the following horizontal loading scale (3) and is so repeated to the lowest scale (4).

After calculation of the effect of fuel on the CG position of the aircraft a vertical is lowered to the lower chart (f) until it intersects the horizontal line of gross weight reading of the aircraft (5). From the point of intersection parallel with the slanted vertical lines of the grid of the lower chart line (6) is drawn until it intersects the horizontal scale, on which the CG position of the loaded aircraft is read.

On the chart is shown an example of calculation of CG position by the data indicated below, which are listed in the order of their use during calculation:

- 1) CG position of empty aircraft - 18.05% of MAC;
- 2) weight of empty aircraft - 3515 kg;

3) weight of passengers (75 kg each), not allowing for 10 kg of baggage, permitted free transportation, - 900 kg;

4) baggage weighing 120 kg (with center of application at frame No 14) - 120 kg;

5) crew 2 men (80 kg each) - 160 kg;

6) oil - 70

7) fuel - 485 kg;

8) gross weight of aircraft - 5250 kg;

9) CG position on aircraft - 31.7% of MAC.

Calculation of CG position of aircraft up to 121 series and above this series is produced with the same order, but by other charts, having different scale values on the horizontal loading scales.

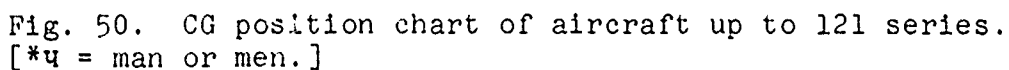
In Fig. 50 there is listed the CG position chart of An-2 aircraft up to 121 series.

When determining the gross weight of the aircraft under winter conditions the weight of one passenger is considered 80 kg, empty aircraft weight with the presence of ski landing gear - 70 kg higher.

When determining the CG position of the aircraft consider passenger seats that due to winter clothes with a full load of passengers the CG position is increased 0.6% of MAC, and due to ski landing gear the CG position is decreased 0.8% of MAC.

Calculation of actual CG position of the aircraft in flight, in case of landing on an airfield with swampy ground, is produced by proceeding from the fact that burnoff of each 100 kg of fuel





decreases the CG position by 0.3% of MAC, and shift of a load weighing 80 kg to frame No 15 increases the CG position by 2.5% of MAC.

### 3. Axes of Rotation of the Aircraft and Moments of Forces Affecting the Aircraft

The position of the aircraft in space is oriented relative to three imaginary axes of the aircraft (Fig. 51): - longitudinal - X; lateral - Z; vertical - Y.

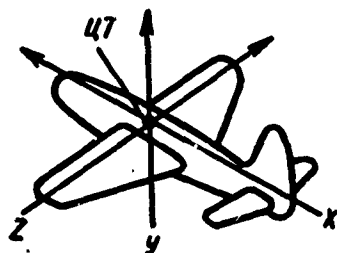


Fig. 51. Axes of rotation of aircraft.

It is accepted to consider that all three axes intersect mutually perpendicularly at the center of gravity of the aircraft.

Forces affecting the aircraft relative to the CG create moments, which change the position of the aircraft relative to its axes of rotation.

Moments, tending to turn the aircraft around its lateral axis, are called longitudinal or pitching moments ( $M_z$ ), since they change the pitch angle.

The pitch angle is the angle concluded between the longitudinal axis of the aircraft and sky line. Pitching moments are subdivided into negative (decreasing the pitch angle) and positive (increasing the pitch angle).

Moments tending to turn the aircraft around its longitudinal axis are called rolling ( $M_x$ ), since they change the value of the angle of roll.

Angle of roll is the angle concluded between the lateral axis of the aircraft and the skyline.

Moments tending to turn the aircraft around the vertical axis ( $M_y$ ) are called yawing moments.

Depending upon the relationship of moments affecting the aircraft, we examine three types of balance, stability and controllability of an aircraft.

#### 4. Concept of Aircraft Balance

The state of an aircraft in flight, at which the center of gravity moves rectilinearly and evenly, and the aircraft does not revolve around its axes is called aircraft balance.

So that the center of gravity of an aircraft would shift rectilinearly and evenly, there is necessary a condition that the sum of all forces affecting the aircraft be equal to zero ( $\Sigma F = 0$ ).

So that the aircraft does not revolve relative to the center of gravity, the conditions is necessary that the sum of all moments affecting the aircraft be equal to zero ( $\Sigma M_{ut} = 0$ ).

Subsequently, considering that equilibrium of forces is attained, only the ratio of moments will be considered.

#### 5. Longitudinal Equilibrium of Aircraft

The state of an aircraft in flight, at which it does not change its position relative to the lateral axis, is called longitudinal equilibrium.

For longitudinal equilibrium of an aircraft there is necessary the condition that pitching moments be mutually balances ( $\Sigma M_z = 0$ ).

Figure 52 shows the ratio of pitching moments at longitudinal equilibrium of An-2 aircraft with neutral position of the elevator, where

$$M_{z_{sp}} = M_{z_p} + M_{z_{r.o.}}$$

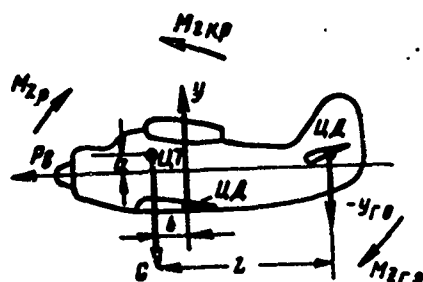


Fig. 52. Longitudinal equilibrium conditions of aircraft in flight.

Longitudinal equilibrium of an aircraft can be disturbed by change of engine power rating, influence of gusts of bumpy air on the aircraft, elevator deflection, change of CG position of aircraft, and deflection of flaps.

Figure 53a shows the ratio of pitching moments in flight during engine operation at cruise setting, at which the power is 0.5 the rated power of the engine, during which the positive pitching moment of thrust ( $M_{z_p}$ ) and horizontal tail surfaces ( $M_{z_{r.o.}}$ ) are balanced by the negative pitching moment of the wing ( $M_{z_{sp}}$ ), the aircraft is in longitudinal equilibrium.

With increase of engine power rating to nominal (Fig. 52b) the negative pitching moment of the wing ( $M_{z_{sp}}$ ) is increased due to increase of airflow ( $V_{\infty 1} > V_{\infty 2}$ ) and vertical component of thrust ( $P_{y1}$ ). The positive pitching moments of thrust ( $P_{Q1} > P_Q$ ) and horizontal tail surfaces ( $-Y_{r.o.1} > -Y_{r.o.}$ ) are increased simultaneously. Increase of  $-Y_{r.o.1}$  is caused by increase of  $-\alpha_{r.o.}$  because of increase of downwash on the horizontal tail surfaces, since  $V_{\infty 1} > V_{\infty 2}$ . With increase of engine power rating the positive pitching moments predominate over the negative pitching moments of the wing ( $M_{z_{p1}} + M_{z_{r.o.1}} > M_{z_{sp1}}$ ), as a result of which the aircraft pitches up.

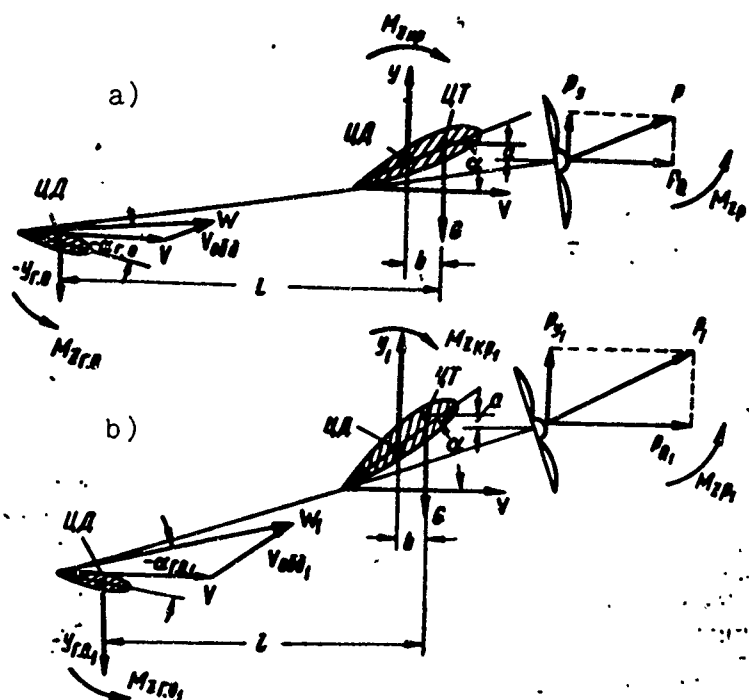


Fig. 53. Effect of engine power rating on aircraft longitudinal equilibrium: a) engine power is equal to 0.5 nominal ( $M_{sp} + M_{sr,0} = M_{sr,p}$  - aircraft in longitudinal equilibrium); b) engine power is equal to nominal ( $M_{sp} + M_{sr,0} > M_{sr,p}$  - aircraft pitches up).

With the influence of gusts of wind on the aircraft the amount of lift is changed, the former equality of pitching moments is disturbed, and aircraft goes out of equilibrium.

With change of CG position in flight due to change of the value of arms, the equality of pitching moments is disturbed; with decrease of CG position the negative pitching moment predominates, increase of CG position - positive.

Figure 54a shows the ratio of pitching moments, at which an aircraft in flight with undeflected flaps is in longitudinal equilibrium. With deflection of flaps (Fig. 54b) due to increase of wing lift ( $Y_1 > Y$ ) and displacement of CP to the trailing edge of the wing ( $b_1 > b$ ) the negative pitching moment of the wing is increased ( $M_{z_{sp1}} > M_{z_{sp}}$ ). Simultaneously with deflection of flaps, due to increase

of downwash,  $-\alpha_{r,0}$  ( $-\alpha_{r,0} > -\alpha_{r,0}$ ), is decreased, which is accompanied by increase of lift, having negative value ( $-Y_{r,0} > -Y_{r,0}$ ). Inasmuch as arm of action  $-Y_{r,0}$  is greater than the arm of action  $Y_{np}$  ( $l > b$ ), the positive pitching moment of horizontal tail surfaces ( $M_{z_{r,0}}$ ) predominates over the increased negative pitching moment of the wing ( $M_{z_{np}}$ ) and the aircraft pitches up with deflection of flaps.

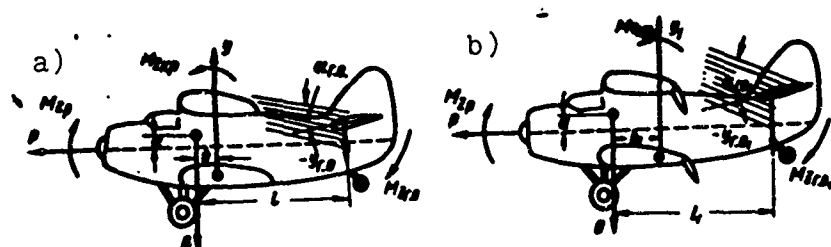


Fig. 54. Effect of deflection of flaps on the longitudinal equilibrium of aircraft: a) ratio of pitching moments before deflection of flaps; b) ratio of pitching moments after deflection of flaps.

Longitudinal equilibrium of aircraft in flight is provided by elevator deflection.

## 6. Concept of Aircraft Stability

The capacity of an aircraft to restore the initial state of equilibrium in flight without interference of the pilot is called stability. According to this capability the aircraft are subdivided into stable, unstable and neutral (being in a state of neutral equilibrium).

Stable aircraft after cessation of the action of external forces, causing disturbance of equilibrium, tends to return to initial state of equilibrium. Unstable aircraft tends to depart still further from the former state of equilibrium.

Neutral aircraft after cessation of the action of external forces remains at the angle of attack into which it was guided by external forces, not tending to return to the initial state of

equilibrium or depart further from it. Flight on an unstable aircraft is possible but is conjugate with large expenditure of physical effort of the pilot, since it requires increased attention and continuous interference for restoration of the disturbed equilibrium.

Flight on a neutral aircraft also requires constant attention of the pilot for maintaining assigned flight conditions, but to a smaller extent than on an unstable.

Moments of forces, tending to return the aircraft to initial state of equilibrium, are called stabilizing. Moments of forces, favoring further withdrawal of the aircraft from state of equilibrium, are called destabilizing.

For easing the analysis of aircraft stability it is artificially divided into static and dynamic.

Static stability characterizes the capability of an aircraft to create stabilizing moments at the very beginning of disturbance of equilibrium and does not characterize, what the movement (tendency) of the aircraft will be with the passage of time after cessation of the action of external forces.

Dynamic stability - this is the capability of the aircraft to restore initial flight conditions without interference of the pilot a certain time after cessation of the action of external forces. Dynamic stability characterizes the entire process of movement of the aircraft to initial state of equilibrium taking into account forces of inertia of the aircraft.

Static stability is a necessary condition of dynamic stability. For all practical purposes an aircraft possessing static stability will also be dynamically stable.

## 7. Longitudinal Stability of Aircraft

Capability of an aircraft in flight without interference of the pilot to restore disturbed longitudinal equilibrium is called longitudinal stability. Inasmuch as the angle of attack and speed with disturbance of longitudinal equilibrium have a different character of change, in aerodynamics we consider angle of attack (overload) and speed longitudinal stability.

Overload ( $n$ ) is the number, showing how many times lift is greater than weight of the aircraft. Due to increase of the angle of attack with disturbance of longitudinal equilibrium, lift will obtain increment ( $\Delta Y$ ), thanks to which the overload will be changed as compared to initial flight conditions. The aircraft is called overload stable if it independently (without interference of the pilot) tends to maintain the overload of initial conditions (to maintain initial angle of attack at constant speed); in other words, the aircraft is overload stable when change of the angle of attack at constant flight speed causes the appearance of stabilizing moment, tending to eliminate the change of angle of attack.

Overload stability shows how the aircraft behaves in the initial moment after disturbance of equilibrium, when the pilot has not yet interfered in aircraft control for restoration of the disturbed longitudinal equilibrium.

The aircraft is called speed stable if it independently, without interference of the pilot, tends to maintain the speed of initial flight conditions.

Let us assume that an aircraft in longitudinal equilibrium under action of a gust of wind changed its flight path, having been deflected upwards. With this the required thrust increases. But since the engine power rating was not changed, the amount of available thrust remained as before. Therefore, the flight speed will start to decrease. Deceleration will lead to decrease of lift, as a result



of which the flight path is curved downwards, aircraft speed is increased to initial value.

Aircraft is speed stable when change of the angle of attack, having occurred as a result of change of flight speed at constant overload, causes the appearance of stabilizing moment, tending to eliminate this change of angle of attack.

For understanding the physical essence of longitudinal stability of an aircraft there are used concepts of aerodynamic centers of wing and aircraft.

#### Aerodynamic Center of Wing and Aircraft

Figure 55a shows that at angle of attack ( $\alpha$ ) the wing creates lift ( $Y$ ), which relative to point  $\Phi$  creates a moment. With increase of angle of attack to  $\alpha_1$  (Fig. 55b) lift ( $Y_1$ ) is increased, and  $CP_1$  of wing of asymmetric profile is shifted forward, as a result of which the arm (from CP to point  $\Phi$ ) is decreased.

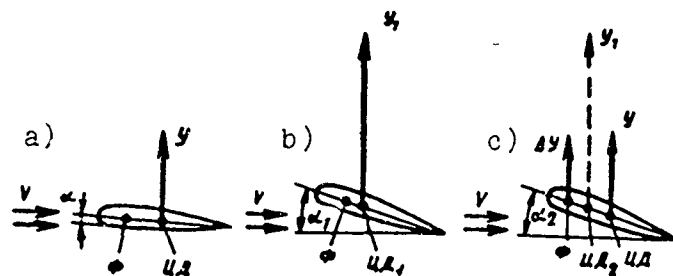


Fig. 55. Aerodynamic center of wing.

It was established that on the chord of the wing it is possible to calculate point  $\Phi$  so that relative to it the amount of lift will be changed inversely proportional to the arm, and then the moment of the wing will not be changed relative to this point with change of angle of attack (Fig. 55c).

Point of application of increment of lift ( $\Delta Y$ ), relative to which the wing moment is not changed with change of angle of attack

is called the aerodynamic center of the wing. Force of increment can be positive (+) with increase of angle of attack and negative (-) with decrease of angle of attack.

Force  $Y_1 = Y + \Delta Y$ , applied at  $CP_1$ , can be expanded into  $Y$ , applied at  $CP$ , and force  $\Delta Y$ , applied at aerodynamic center (see Fig. 55c). Since moment of force  $\Delta Y$  relative to point " $\phi$ " is equal to zero, then the wing moment at angle of attack  $\alpha_1$  will be the same as at angle of attack  $\alpha$ . For the majority of wing profiles the aerodynamic center is at distance of 23-25% from the leading edge of the chord.

Horizontal tail surfaces, just as the wing, have their aerodynamic center. With change of angle of attack there appear increments of lift both on wing and on horizontal tail surfaces, which will be applied at aerodynamic centers of wing and horizontal tail surfaces respectively (Fig. 56).

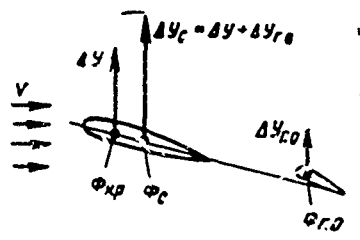


Fig. 56. Aerodynamic center of aircraft.

Point of application of resultant increments of lifts  $\Delta Y$  and  $\Delta Y_{c0}$  is called the aerodynamic center of aircraft. According to rules of mechanics the aerodynamic center will be at a distance inversely proportional to values of increment of forces  $\Delta Y$  and  $\Delta Y_{c0}$ .

Since increment of lift on the wing is always greater than on horizontal tail surfaces (because of larger area), the aerodynamic center of aircraft will always be on the wing chord.

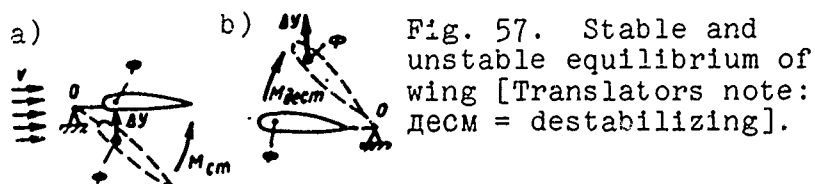
#### Condition of Longitudinal Stability of Isolated Wing

If we secure a model on a hinged support and set it in the flow of a wind tunnel, balanced at a certain angle of attack, as is shown by the solid line in Fig. 57, then it is possible to establish:

1) with increase of angle of attack (position of wing is shown by dotted line) the lift will be increased by value  $\Delta Y$ ;

2) if the hinge, which is the center of gravity on the aircraft, is located ahead of the aerodynamic center of the wing (Fig. 57a), then force  $\Delta Y$  creates stabilizing moment.

3) if the hinge is behind the aerodynamic center of the wing (Fig. 57b), force  $\Delta Y$  creates destabilizing moment.



From the considered example the conclusion can be made that a condition of longitudinal stability of an isolated wing is the mutual location of aerodynamic center of the wing and center of gravity of the aircraft.

#### Condition of Longitudinal Stability of Aircraft

Just as for an isolated wing, the longitudinal stability of aircraft is determined by the mutual location of center of gravity and aerodynamic center of the aircraft.

Figure 58a shows that with location of center of gravity behind the aerodynamic center the aircraft is unstable, since with increase of angle of attack (Fig. 58b) force  $\Delta Y$  relative to the center of gravity creates destabilizing positive pitching moment, and with decrease of angle of attack (Fig. 58c) – destabilizing negative pitching moment.

With location of aerodynamic center of aircraft behind the center of gravity, as shown in Fig. 59a, the aircraft is stable.

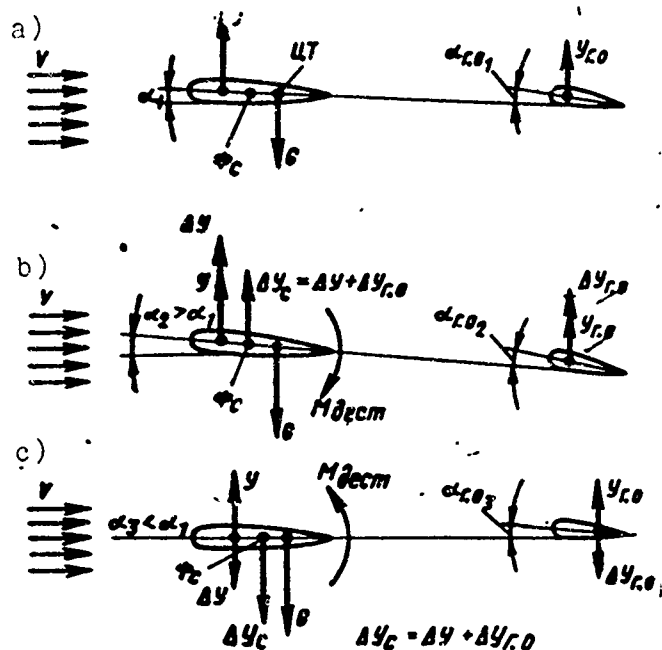


Fig. 58. Condition of longitudinal stability of aircraft, when the aerodynamic center is located ahead of the center of gravity: a) initial distribution of forces; b) distribution of forces and moments with increase of angle of attack; c) distribution of forces and moments with decrease of angle of attack.

With increase of angle of attack (Fig. 59b) force  $\Delta Y_c$  creates stabilizing negative pitching moment, and with decrease of angle of attack – stabilizing positive pitching moment (Fig. 59c). In the case when the center of gravity is combined with the aerodynamic center, the aircraft is in a state of neutral equilibrium.

CG position of aircraft, corresponding to combination of center of gravity with aerodynamic center of aircraft, is called neutral or critical. The difference between neutral and critical CG positions, expressed in percent of mean aerodynamic chord of the wing, is called CG position margin or stability margin.

From the considered examples there can be made conclusions:

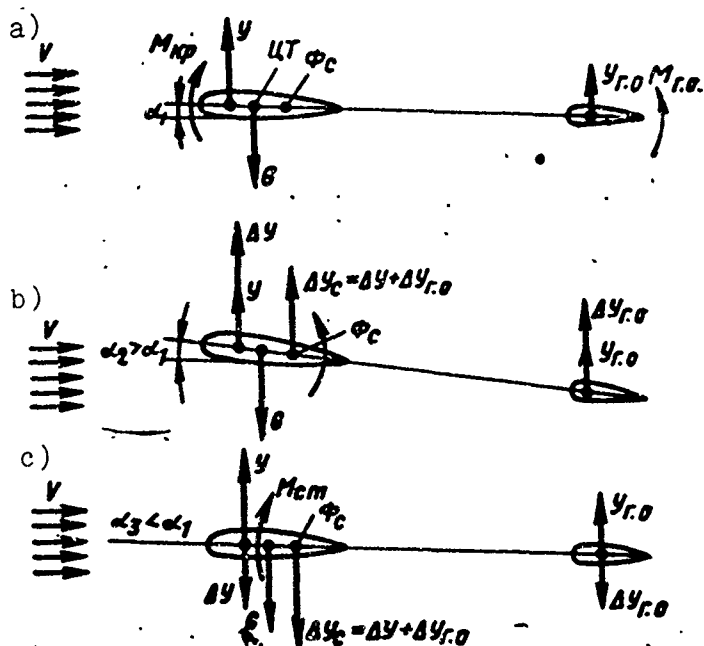


Fig. 59. Conditions of longitudinal stability of aircraft, when the aerodynamic center is located behind the center of gravity: a) initial flight conditions; b) with increase of angle of attack; c) with decrease of angle of attack.

1) in all cases the additional moment of horizontal tail surfaces is stabilizing;

2) depending upon aircraft CG position the additional wing moment can be either destabilizing (at aft CG position), or stabilizing (at forward CG position).

An-2 aircraft under basic flight conditions in the range of established CG position (17.2-33% MAC) has a large overload longitudinal stability margin.

#### Factors Affecting Longitudinal Stability of Aircraft

The longitudinal stability of aircraft is affected by those factors which can lead to change of the distance between the aerodynamic center of aircraft and center of gravity, namely: CG position

of aircraft, engine power rating, deflection of flaps, and also installation of agricultural equipment on the aircraft and, besides this, the flight altitude.

Effect of CG position. With decrease of CG position the arm from the aerodynamic center to the center of gravity is increased, and consequently, also the stabilizing moment. Therefore, longitudinal stability of the aircraft is improved.

Effect of power system. With disturbance of equilibrium the increment of lift at the horizontal tail surfaces will be insignificant since with turn of the aircraft around the lateral axis the angle of attack of horizontal tail surfaces will scarcely be changed during engine operation (it is increased due to turning of the aircraft as much as it is decreased due to increase of downwash).

Besides this, with turning of the aircraft relative to the lateral axis in the direction of increase of pitch angle, the vertical component of thrust ( $P_y$ ) is increased, which will lead to increase of force  $\Delta Y$  and will cause displacement of the aerodynamic center of aircraft forward. Therefore, longitudinal stability worsens.

At the same angle of attack the aircraft stability in a climb will be considerably worse than in a glide.

Effect of flight altitude. The higher the flight altitude, the less the air density, consequently, the less the increment of lift is. Therefore, with climb to altitude the longitudinal stability of the aircraft worsens.

Effect of deflection of flaps. With deflection of flaps the amount of lift increment of the wing is increased. In this case there is simultaneously increased downwash in the region of horizontal tail surfaces, the aerodynamic center of the aircraft is displaced forward, and longitudinal stability worsens.

Impairment of longitudinal stability of an aircraft with deflection of flaps is also explained by the fact that increment of lift on horizontal tail surfaces is decreased because of deceleration of flow rate by flaps.

Effect of installation of agricultural equipment. With installation of agricultural equipment on the aircraft its longitudinal stability worsens because in this case the horizontal tail surfaces are streamlined more by the airstream, the speed of which is also decreased because of its deceleration by the agricultural equipment.

Comparatively large overload longitudinal stability margin of An-2 aircraft in the range of established CG positions at basic conditions of flight is provided by the large area of horizontal tail surfaces, aspect ratio of rear section of fuselage, and high location of horizontal tail surfaces.

With increase of area of horizontal tail surfaces the increment of lift ( $\Delta Y_{r.0}$ ) at other identical conditions is increased, aerodynamic center of the aircraft is displaced closer to the horizontal tail surfaces, therefore, longitudinal stability of the aircraft is improved. With increase of length of rear section of the fuselage the arm of action of force  $\Delta Y_{r.0}$  is increased, therefore, longitudinal stability is improved.

With high location of horizontal tail surfaces the airstream, trailing from the wing, passes below it, increment of lift on horizontal tail surfaces is increased, aerodynamic center of aircraft is displaced aft, increasing the stabilizing moment.

Limiting aft CG position (33% MAC) is established for the purpose of providing the aircraft a sufficient longitudinal stability margin in flight at nominal engine power with undeflected flaps.

## 8. Concept of Aircraft Controllability and Requirements Imposed on It

Capability of the aircraft to change flight conditions with corresponding change of the position of aircraft control vane by the pilot is called controllability.

A well controlled aircraft should correspond to the following requirements:

1. Have perceptible amount of load on flight controls of the aircraft.
2. Have average deflection of controls during transition from one flight condition to another. Thus, for example, for changing the gliding speed of An-2 aircraft from 120 to 150 km/h it is required to deflect the elevator downwards; with CG position 35.6% MAC — by  $1^\circ$ , with CG position 24.1% MAC — by  $4^\circ$ , and with CG position 31.3% MAC — by  $2^\circ$ .

From this it is seen that the aircraft will have the best controllability at CG position 31.3% MAC, since deflection of controls in this case is  $2^\circ$ . Too much and too little deflection of controls are equally unacceptable, since they fatigue the pilot.

3. Have control deflection margin unusable in normal flight in case of an emergency situation: spontaneous shift of cargo during flight, severe loss of speed, malfunction in the construction. In climb at nominal engine power with deflected flaps at CG position 31.3% MAC for longitudinal balance of An-2 aircraft at speed 120 km/h there is required  $13^\circ$  elevator deflection downwards.

Inasmuch as the critical angle of elevator deflection downward of an An-2 aircraft is  $22^\circ$ , the elevator deflection margin in this case will be  $9^\circ$ , or 40%.



4. Possess harmony: the stronger and sharper the action of the pilot, the more energetically the aircraft should react.

5. Rapidly perceive the action of controls, i.e., not allow delay in change of its position with deflection of controls.

6. Have naturalness of movement of controls: deflection of control column forward should be accompanied by increase of flight speed, aft - deceleration of flight speed.

Principle of action of controls and ailerons is similar and is based on change of wing profile camber or tail surface curvature of aircraft. During elevator deflection there is changed the profile curvature and character of airflow of horizontal tail surfaces; change of character of airflow causes change of distribution of pressure along the profile of tail surfaces, and consequently, the amount and direction of aerodynamic force of horizontal tail surfaces (Fig. 60).

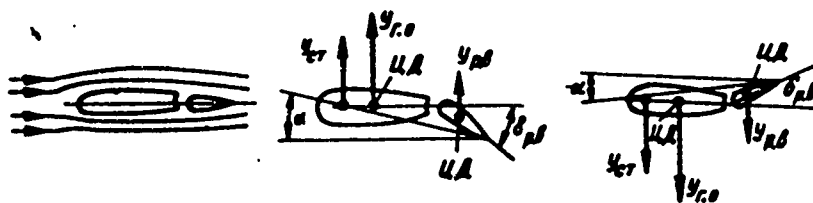


Fig. 60. Principle of action of elevator.

From the figure it is clear that the stabilizer and elevator participate in creation of lift of horizontal tail surfaces, but change of  $Y_{\text{r.o}}$  occurs basically due to the stabilizer, since its area and thickness of profile are considerably greater than at the controls. Aerodynamic force of horizontal tail surfaces, created during elevator deflection, is directed to the side opposite deflection of control.

## 9. Methods of Decrease of Forces on Aircraft Flight Controls

While controlling the aircraft, the pilot deflects the controls and ailerons, applying certain forces ( $P$ ) for this. The amount of these forces is determined by the product of aerodynamic force of elevator ( $Y_{p.s.}$ ) by arm ( $a$ ) from the center of pressure to the axis of rotation of the control, which is called hinge moment ( $M_m = Y_{p.s.} \cdot a$ ) - Fig. 61.

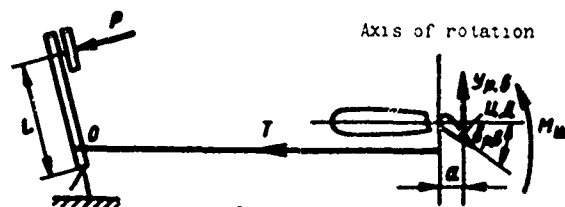


Fig. 61. Determination of force on aircraft flight controls.

The value of hinge moment will be greater, the greater the area, control deflection angle and flight speed will be, since on these factors depends the amount of aerodynamic force of the control.

For decrease of hinge moment of controls and ailerons on the aircraft there are applied aerodynamic balance and trim tabs (Fig. 62).

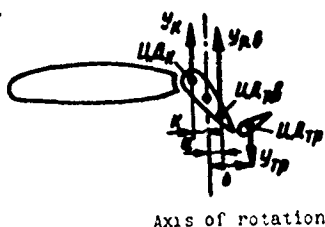


Fig. 62. Principle of action of aerodynamic balance and trim tab.

Aerodynamic balancing of control surfaces is the part of the area of control surface located in front of its axis of rotation. Aerodynamic balance is: ailerons - 21.7%, elevator - 24%, rudder - 19%.

Trim tab - this is a plate, hinged close to the trailing edge of control surface and controlled in flight at the pilot's will.

With deflection of control surface on the balance appears aerodynamic force  $Y_k$ , which relative to axis of rotation of control surface (arm  $k$ ) creates moment ( $\dot{M}_k = Y_k \cdot a$ ), directed to the opposite side of hinged moment of control surface, and, thus, decreases it.

By changing the trim tab deflection angle, the pilot changes the value of moment ( $M_{tp} = Y_{tp} \cdot b$ ), and consequently, the value of hinged moment of control surface. With the help of the trim tab it is possible to decrease the force on the lever to any value, right up to complete removal of forces ( $M_w = M_k + M_{tp}$ ).

#### 10. Longitudinal Controllability of Aircraft

Capability of the aircraft to change its position relative to the lateral axis with deflection of elevator by the pilot is called longitudinal controllability.

Longitudinal controllability is characterized by degree of controllability, i.e., amount of change of angle of attack with elevator deflection  $1^\circ$ .

According to degree of controllability ( $\epsilon$ ) the aircraft are subdivided into three groups:

- a) insufficiently controllable -  $\epsilon$  up to 1;
- b) well controllable -  $\epsilon = 1.5-2$ ;
- c) unnecessarily controllable -  $\epsilon$  over 3.

Degree of aircraft controllability depends on the engine power rating, flight speed and CG position of aircraft, with increase of which it is increased.

# 11. Balance Curves, Analysis of Longitudinal Controllability of Aircraft by Them

Charts, showing what amount it is necessary to deflect the elevator for longitudinal balance of the aircraft depending on the flight speed, engine power rating, CG position of aircraft and position of flaps, are called balance curves. Balance curves of An-2 aircraft are composed by State Scientific Research Institute of Civil Aviation (GosNIIGA) according to flight tests of aircraft: in gliding with undeflected flaps, in flight at nominal engine power with undeflected flaps and at nominal engine power with flaps deflected 30°.

Balance curves are listed for CG positions 24.1, 31.3, and 35.6% MAC. Each chart is represented by a system of coordinates, along the vertical axis of which are plotted elevator deflection angles: downwards with plus sign (+), upwards with minus sign (-); along the horizontal axis is plotted indicated flying speed. Each curve corresponds to a specific CG position of the aircraft and shows the required amount of elevator deflection depending upon the shown factors.

When gliding with undeflected flaps (Fig. 63) the aircraft with CG position 24.1% MAC at speed 118 km/h is in longitudinal equilibrium with neutral position of elevator. The flight speed, at which for longitudinal balance of the aircraft it is not required to deflect the elevator, is called equilibrium speed.

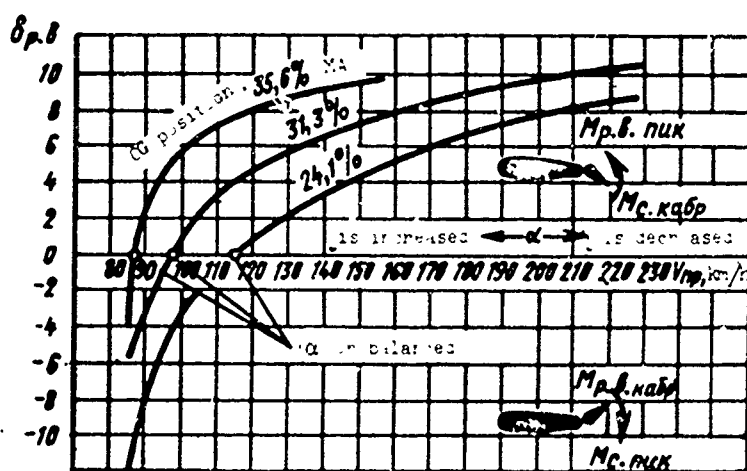


Fig. 63. Balance curves of aircraft during gliding with undeflected flaps. [Translators note: ПИК = negative pitching; кабр = positive pitching.]

When gliding at speeds over 118 km/h on the aircraft the positive pitching moment is predominant, for balance of which there is required elevator deflection downwards, with increase of flight speed the amount of balance elevator deflection is increased. When gliding at speeds less than 118 km/h on the aircraft the negative pitching moment is predominant, for balance of which there is required elevator deflection upwards.

In proportion to deceleration of flight speed the amount of balance elevator deflection is increased, which is caused by impairment of its effectiveness at lower speeds. During aircraft flight at nominal engine power with undeflected flaps (Fig. 64) positive pitching moment is predominant on the aircraft. For aircraft balance there is required elevator deflection downwards, which is increased with increase of flight speed.

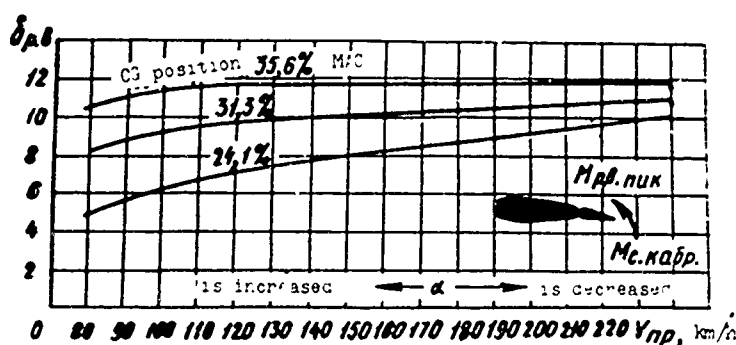


Fig. 64. Balance curves of aircraft during flight at nominal power with undeflected flaps.

During aircraft flight at nominal engine power with flaps deflected 30° (Fig. 65), because of additional increase of downwash on the horizontal tail surfaces the positive pitching moment of the aircraft is increased, which requires increase of balance elevator deflection downwards.

With increase of CG position of aircraft at all flight conditions the amount of positive pitching moment is increased, since due to decrease of the arm from center of gravity to the center of pressure

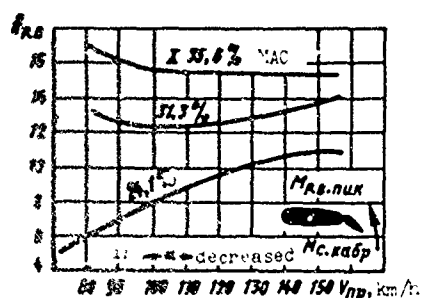


Fig. 65. Balance curves of aircraft during flight at nominal power with flaps deflected  $30^\circ$ .

the negative pitching moment of the wing is decreased. During aircraft flight at nominal engine power with deflected flaps  $30^\circ$  at CG position  $1.3\%$  MAC and greater, with deceleration to  $110$  km/h the positive pitching moment is decreased, which is caused by decrease of downwash angle on the horizontal tail surfaces. With further deceleration (increase of angle of attack) the positive pitching moment is increased, which is explained by increase of the vertical component of thrust ( $P_y$ ) and decrease of the real angle of attack of horizontal tail surfaces by propeller slipstream.

The balance curves represented in Figures 63-65 permit analyzing the longitudinal controllability, the degree of which is judged by slope angle of balance curves to the axis of speeds.

On balance curve of aircraft with CG position  $35.6\%$  MAC when gliding with undeflected flaps (see Fig. 62) one may see that:

- 1) flying speed of aircraft  $88$  km/h is equilibrium;
- 2) at flight speed  $128$  km/h the balance elevator deflection downwards is  $8.5^\circ$ , and at flight speed  $168$  km/h  $-10.3^\circ$ ; with change of flight speed by an identical value ( $40$  km/h) in the second case there was required elevator deflection  $1.8^\circ$  ( $10.3-8.5^\circ$ ) in all, which indicates that longitudinal controllability of aircraft in the speed range  $128-168$  km/h is considerably better than in speed range  $88-128$  km/h;
- 3) in speed range  $88-128$  km/h, where controllability is worse, angle of inclination of balance curve to the axis of speeds is

considerably greater than in speed range 128-168 km/h, where controllability is better.

Thus, the greater the angle of inclination of balance curve, proceeding from bottom to top to the right the worse the aircraft controllability.

By analyzing balance curves, the following conclusions can be made about longitudinal controllability of An-2 aircraft.

1. In glide conditions with undeflected flaps the slope of curves is increased and longitudinal controllability of the aircraft worsens in proportion to deceleration and decrease of CG position of aircraft (see Fig. 63). For balance of aircraft at speed 88 km/h (close to landing speed) with CG position 31.3% MAC elevator deflection is above  $6^\circ$ , whereas with CG position 24.1% MAC for balance of aircraft at the given speed there is required  $12^\circ$  elevator deflection. Consequently, with considerable decrease of CG position and full elevator deflection upwards the moment of lift of horizontal tail surfaces can turn out to be insufficient for creation of landing angle of attack for the aircraft.

To assure sufficient longitudinal controllability of aircraft under gliding conditions and during landing there is established a forward CG limit (17.2% MAC).

2. During aircraft flight at nominal engine power with undeflected flaps, as compared to gliding, the slope of balance curves is decreased degree of longitudinal controllability of aircraft is increased (Fig. 64) more considerably, the greater the CG position of aircraft and the flight speed.

During flight at over 120 km/h with neutral CG position (35.6% MAC) the aircraft is in state of neutral equilibrium and has unnecessary controllability (course of balance curve is parallel to axis of speeds). So that the aircraft would be normally controllable during flight at nominal engine power with undeflected flaps, there is established CG limit 33% of MAC.

3. During aircraft flight at nominal engine power with flaps deflected  $30^\circ$  (Fig. 65), and all the more so during use of takeoff power and deflection of flaps at  $40^\circ$  angle, at flight speeds less than 95 km/h with CG position of aircraft 31.3% MAC and greater, the longitudinal controllability of aircraft is sharply impaired: excess positive pitching moment is predominate on aircraft, it is increased in proportion to deceleration, and elevator deflection margin is reduced. However, even with CG position 35.6% MAC the deflection margin of control surface composes  $7^\circ$ , or 30%.

With CG position of aircraft 24.1% MAC at the entire speed range of flight and with CG position 31.3% at flight speed over 100 km/h the longitudinal controllability is impaired insignificantly as compared to flight of aircraft at nominal power with undeflected flaps (difference in balance elevator deflection downwards is approximately  $2^\circ$  greater).

For the purpose of preventing loss of controllability of aircraft it is necessary to:

1) correctly accomplish loading of aircraft and reliably secure cargo in the aircraft; after distribution of cargo thoroughly check the CG position of aircraft by the CG position chart and do not permit flight at CG position exceeding limiting aft CG (33% of MAC);

2) before takeoff set elevator trim tab in the position corresponding to 2-3 forward depressions of the pressure switch;

3) when performing a landing, as a rule, deflect flaps up to  $30^\circ$  and only in exceptional cases (during short-field landing site) use flaps with  $40^\circ$  deflection angle;

4) make a decision on balked landing ahead of time; do not permit sharp increase of engine power, and parry a rising positive pitching moment with increase of power by smooth forward deflection of control column;



5) since for a long time the indicated airspeed of aircraft does not change, whereas the pitch angle is increased, then with balked landing and with shift of aircraft to takeoff acceleration stage, flight control conditions by the position of aircraft relative to the horizon;

6) in case of loss of controllability during takeoff and balked landing, which can occur only at an excessively aft CG position, decrease engine power until cessation of pitching of aircraft and with the presence of vertical velocity climb 50 m in 2-3 procedures, retract flaps, and then set cruising climb performance.

## 12. Transverse Equilibrium of Aircraft

The state of an aircraft, at which it does not change its position relative to longitudinal axis, is called transverse equilibrium. A condition of transverse equilibrium is equality of rolling moments ( $M_{п.кР} = M_{л.кР}$ ) - Fig. 66.

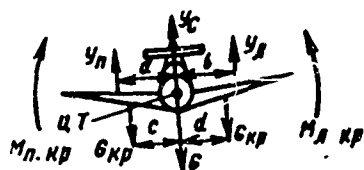


Fig. 66. Diagram of forces and moments, affecting the aircraft relative to the longitudinal axis.

Transverse equilibrium is provided by weight and geometric symmetry of the aircraft and can be disturbed by:

a) change of engine power rating by means of change of value of reactive moment of propeller;

b) nonuniform distribution of load along the lateral axis, which is characteristic for An-2 aircraft, since with position of four-way fuel cock at "Tanks open" there occurs nonuniform fuel burnoff (faster from the left group of tanks than from the right);

c) change of angles of attack on wing half-cells under the influence of vertical airflows or deflection of ailerons;

d) putting the aircraft into slip under the influence of a lateral gust of wind or rudder deflection. Disturbed equilibrium of transverse moments is restored by deflection of ailerons.

### 13. Lateral Stability of Aircraft

Capability of the aircraft to restore initial position of transverse equilibrium in flight without interference of the pilot is called lateral stability.

Figure 67 shows that with disturbance of transverse equilibrium under the influence of vertical flow ( $u$ ) the angle of attack of lowering half-cell of wings ( $\alpha_{on}$ ) is increased, and the raising half-cell ( $\alpha_{nox}$ ) is decreased, which creates a difference in value of lifts on wing half-cells ( $Y_{on} > Y_{nox}$ ). Resultant lift ( $Y$ ) is displaced in the direction of the lowered half-cell of wings and, acting on the arm up to  $CG \cdot a$ , creates damping (braking) moment ( $M_{\text{демп}}$ ), preventing further increase of angle of roll. However, damping moment cannot restore transverse equilibrium of the aircraft, since it acts only in the process of disturbance of equilibrium, and then ceases its action.

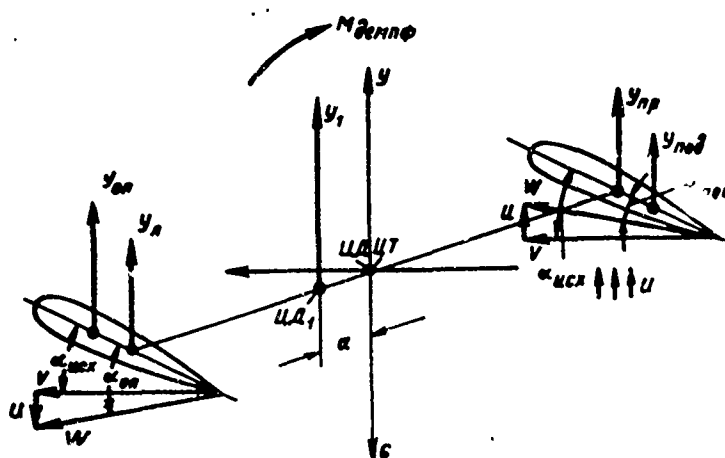


Fig. 67. Damping moment during disturbance of transverse equilibrium (designation of forces without subscripts corresponds to initial flight conditions, with subscript - in the process of disturbance of equilibrium).

Transverse equilibrium of aircraft is restored by the stabilizing moment appearing during aircraft slip on the lowered wing under action of force  $Z_{ck}$  - resultant force of weight (G) and lift (Y), which is shown in Fig. 68a.

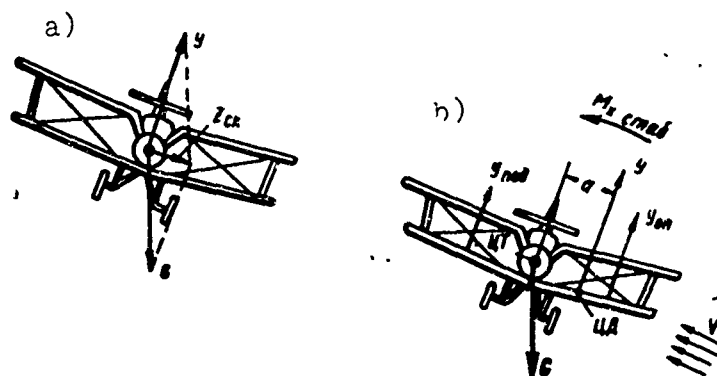


Fig. 68. Appearance of transverse stabilizing moment during aircraft slip: a) formation of force; b) appearance of transverse stabilizing moment.

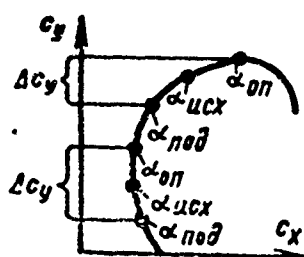
During slip on the lowered half-cell of wings because of aerodynamic blanketing of the raised half-cell of wings by the fuselage, on the lowered wing half-cell there is created a greater amount of lift ( $Y_{on}$ ), than on the raised ( $Y_{nob}$ ).

Resultant lift (Y), as is shown in Fig. 68b, is displaced in the direction of lowered half-cell of wings and, acting on the arm relative to the center of gravity (a), creates stabilizing moment ( $M_{x_{cmsb}}$ ), which after cessation of the action of external forces (u - in Fig. 66) will restore the initial state of equilibrium.

The amount of damping moment depends on the wing area, with increase of which it is increased, since at constants  $\frac{\rho V^2}{2}$  and  $\Delta\alpha$  on the section of proportional change of  $c_y$  by  $\alpha$  the value of  $\Delta Y$  depends only on the area of lowered wing ( $\Delta Y = \Delta c_y \frac{\rho V^2}{2} S$ ).

The amount of stabilizing moment depends on the dihedral angle and aspect ratio. Figure 69 shows that in the presence of dihedral

With increase of angles of attack of both wing half-cells, because of decrease of the rise of coefficient  $c_y$ ,  $(\Delta Y)$  is decreased, as a result of which lateral stability of the aircraft worsens; at angles of attack close to critical, the aircraft loses the capability to independently restore the initial state of transverse equilibrium, which may be seen in Fig. 70. Lateral stability of An-2 aircraft at large angles of attack is provided by the presence of automatic slats, rectangular shape of wing and thick wing profile.



Release of slat after achievement of  $\alpha = 16^\circ$  by the lowered wing half-cell sharply increases the stabilizing moment. Increase of vertical velocity from the root to the outer wing panel of a

rectangular wing increases the downwash under the wing at its outer panel, by which the angle of attack is decreased, and consequently, flow separation at outriggers is delayed, as a result of which the lateral stability of aircraft is improved. During transition of aircraft to supercritical angles of attack due to thick wing profile the decrease of  $\Delta Y$  occurs slower than for wings with thin profile, which promotes improvement of lateral stability of the aircraft.

#### 14. Transverse Controllability of Aircraft

Capability of the aircraft to change its position in flight relative to the longitudinal axis during deflection of ailerons by the pilot is called transverse controllability. Principle of action of ailerons is analogous to the principle of action of control surfaces. A peculiarity of operation of ailerons is the fact that with deflection of the control wheel the aircraft can continuously revolve around the longitudinal axis, since the damping moment created with this is not in a state to balance the aircraft at a specific angle of bank.

Deflection of ailerons at small angles of attack causes insignificant change of coefficient  $c_x$ , which for all practical purposes does not lead to turning of the aircraft. The aircraft is turned due to centripetal force, appearing during aircraft banking.

In proportion to increase of the angle of attack, transverse controllability of the aircraft worsens, and under certain conditions there can set in a complete loss of controllability. Impairment of controllability at large angles of attack is explained by the slow increase of lift coefficient ( $c_y$ ), as a consequence of which rolling moment  $M_{kp}$  is insignificant. Besides this, at large angles of attack drag at wing with lowered aileron ( $Q_{эл.оп}$ ) due to induced drag is considerably greater than at a wing with raised aileron ( $Q_{эл.под}$ ), which creates turning moment ( $M_p$ ) toward the wing with lowered aileron (Fig. 71).

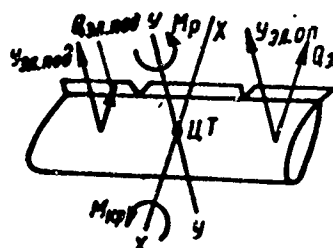


Fig. 71. Appearance of bank and turn of aircraft with deflection of ailerons at large angles of attack.

Thus, with deflection of control wheel the aircraft is slowly banked in the direction of turn of the control wheel (raised aileron) and slips to the opposite side (on the wing with lowered aileron), which impairs transverse controllability. In case of equality of moments, created by deflection of ailerons and appearing because of aircraft slip, loss of controllability advances. Predominance of the moment induced by aircraft slip over the basic rolling moment will lead to reverse controllability: with deflection of control wheel to the left the aircraft will be banked and turn to the right.

Good transverse controllability of An-2 aircraft at large angles of attack is assured by application of slotted ailerons with differential deflection. Principle of operation of slotted ailerons is analogous to that of slotted flaps: they increase the rolling moment in the direction of the raised aileron and decrease the induced drag of wing with lowered aileron, and consequently also the turning moment in the direction of the lowered aileron.

Essence of differential deflection of ailerons (Fig. 72) consists of the fact that with deflection of the control wheel the aileron is deflected upwards at a larger angle ( $30^\circ$ ) than downwards ( $14^\circ$ ), as a result of which the following occurs:

a) on a wing with lowered aileron the lift ( $Y_{эл.оп}$ ) is somewhat decreased and drag ( $Q_{эл.оп}$ ) is considerably decreased;

b) on a wing with raised aileron profile drag ( $Q_{эл.под}$ ) is increased, in view of the fact that part of the aileron exceeds the boundary layer limits.

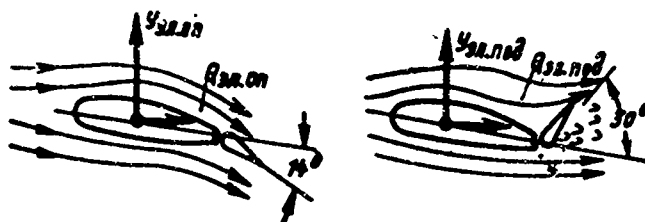


Fig. 72. Differential deflection of ailerons.

Thus, ailerons with differential deflection increase the rolling moment in the direction of the wing with raised aileron and decrease the turning moment in the direction of the wing with lowered aileron. Transverse controllability of An-2 aircraft at large angles of attack is also improved with release of slats. Under conditions of pancaking with retracted slats at a speed of 90-95 km/h the aircraft reacts well to deflection of ailerons and permits performing the maneuver with a bank up to  $15^\circ$ .

#### 15. Directional Equilibrium of Aircraft

The state of an aircraft, at which it does not change its position relative to the vertical axis, is called directional equilibrium. A condition of directional equilibrium is equality of yawing moments ( $M_{yn} = M_{yA}$ ) - Fig. 73.

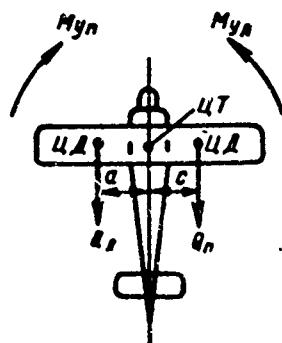


Fig. 73. Diagram of forces and moments, affecting the aircraft relative to directional axis.

Directional equilibrium can be disturbed by:

- a) nonuniform fuel burnoff from the group tank, which will lead

to change of the value of arms of drag forces of wing half-cells;

b) lateral gust of wind or rudder deflection;

c) banking of aircraft, at which centripetal force appears, causing rotation of the aircraft around the vertical axis.

## 16. Directional Stability of Aircraft

Capability of the aircraft to restore the initial state of directional equilibrium without interference of the pilot is called directional stability. With disturbance of directional equilibrium, due to turn of the aircraft around its vertical axis ( $M_{\text{pass}}$ ), on the vertical tail surfaces and fuselage appear lateral aerodynamic forces. The difference of drag forces on wing half-cell ( $Q_n > Q_s$ ) and lateral aerodynamic forces ( $P_{\text{top}}$ ), directed opposite the direction of turn of aircraft (Fig. 74) relative to the center of gravity, create a damping moment, preventing infinite rotation of the aircraft around its vertical axis. In spite of cessation of aircraft rotation under the action of damping moment, it continues motion in the former direction by inertia.

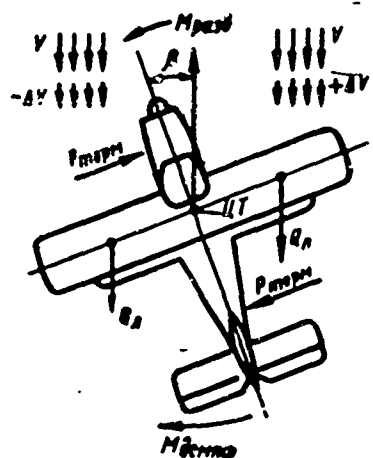


Fig. 74. Damping moment aircraft turn.

During aircraft motion with slip, which is shown in Fig. 75, vertical tail surfaces and the lateral surface of the fuselage create stabilizing moment relative to the center of gravity, under



action of which the aircraft tends to become in the direction of flow ( $M_{CT} = M_{Z_{z.0}} + M_{Z_{z.\phi}} - M_{Z_{x.\phi}}$ ).

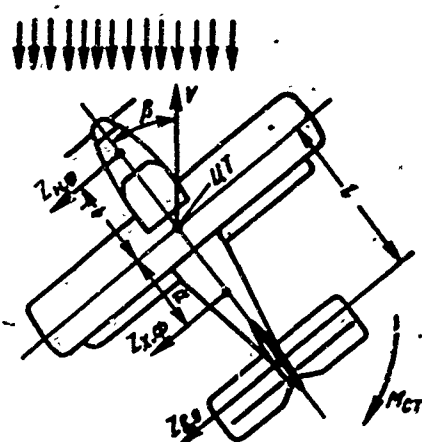


Fig. 75. Stabilizing moment during aircraft flight with slip.

Thus, the amount of stabilizing moment depends on the area of vertical tail surfaces, length of rear section of fuselage and CG position of aircraft. Good directional stability of An-2 aircraft is assured due to large area of vertical tail surfaces, aspect ratio of rear section of fuselage and insignificant length of the fuselage nose, and also establishment of aft CG position limit.

#### 17. Directional Control of Aircraft

Capability of the aircraft to change its position in flight relative to the vertical axis with deflection of the rudder by the pilot is called directional control.

With deflection of the rudder by the pilot the aircraft in first period, under action of force of inertia, will continue motion in the former direction and simultaneously under action of turning moment ( $M_y$  - Fig. 76a) will turn around the vertical axis, as a result of which there is created angle of slip ( $\beta$  - Fig. 76b).

The difference between lateral aerodynamic force ( $Z_\phi$ ), formed during slip, and lateral aerodynamic force of the vertical tail surfaces ( $Z_{z.0}$ ), formed with rudder deflection, will create unbalanced

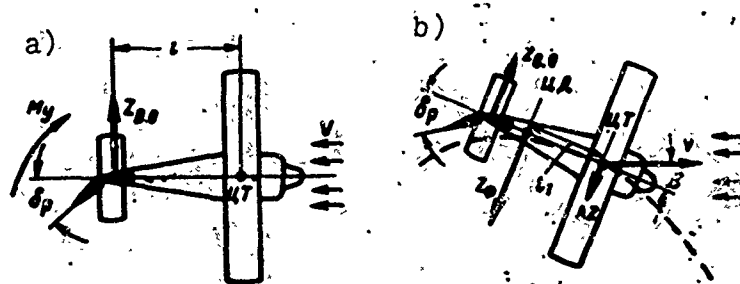


Fig. 76. Principle of directional controllability of aircraft.

centripetal force ( $\Delta Z$ ). Under action of centripetal force the aircraft will turn, moving along a curved flight path, in the direction of rudder deflection. An-2 aircraft has good directional controllability. It is provided by large area of vertical tail surfaces, aspect ratio of rear section of fuselage, large rudder angle ( $28^\circ$  to each side) and establishment of aft CG limit.

#### 18. Transverse Equilibrium of Aircraft

During analysis of the causes of disturbance of transverse and directional equilibrium it was established that with disturbance of transverse equilibrium, due to the creation of centripetal force, there occurs disturbance of directional equilibrium, and with disturbance of directional equilibrium, due to asymmetric airflow of wing half-cell, there occurs disturbance of transverse equilibrium. Such mutual influence of transverse and directional equilibrium on the state of the aircraft is called transverse equilibrium.

#### 19. Lateral Stability of Aircraft

Transverse and directional stability, just as transverse and directional equilibrium, cannot exist separately, their totality is called lateral stability.

For lateral stability of aircraft a condition is necessary so that for the aircraft the angle of roll and angle of slip would be simultaneously removed, i.e., so that the ratio between transverse

and directional stability would be equal to one. With predominance of lateral stability on the aircraft it acquires oscillatory lateral instability, with predominance of directional stability - spiral lateral instability.

An-2 aircraft has good lateral stability, which is attained by correct selection of area of vertical tail surfaces and length of rear section of fuselage, assuring directional stability, and also by value of dihedral angle of wings, area of cell and wing aspect ratio, assuring lateral stability.

## 20. Connection Between Transverse and Directional Controllability

Between transverse and directional controllability there exists the same connection as between transverse and directional equilibrium: bank causes turn of the aircraft, and turn - bank.

However, turn of the aircraft by deflection of only the rudder or only the ailerons occurs with slip. Slip accelerates the flow separation from the wing, which reduces the flight safety and creates additional drag, which requires increase of required thrust. For removal of slip in normal flight the turns are accomplished by coordinated rudder and aileron deflection.

With partial failure of control in flight it is possible to perform a turn for landing approach by deflection of only ailerons alone or the rudder. In this case for safety of flight one should not allow a bank or angle of slip more than  $10^\circ$ , and should maintain speed 5-10 km/h higher than established.

During construction of the route consider that the turning radius will be increased considerably.

## CHAPTER VI

### AIRCRAFT SPIN

Spin occurs when an aircraft descends in a spiral trajectory with simultaneous spontaneous rotation around the longitudinal axis.

Before 1916 it was considered that spin is a random phenomenon, caused by gusts of wind, air "pockets" etc. By observing motion of an aircraft in a spin, the Russian pilot K. K. Artseulov concluded that spin is a fully regular phenomenon. In 1916 Artseulov experimentally unlocked the secret of recovery. However, the physical essence of a spin was still not completely clear, and a method of calculating aircraft on spin, had not been developed and so many aircraft even with correct actions on the part of the pilot did not come out of a spin. The Soviet scientist V. S. Pyshnov theoretically substantiated spin and developed method of calculating aircraft on spin.

#### 1. Causes and Kinds of Spin

To prevent an aircraft from going into a spin, and in case of a stall to ensure the safety of the flight, every pilot is obligated to know clearly the causes of spin and to understand how to recover an aircraft from a spin.

The cause of a spin is loss of controllability because the aircraft goes into critical angles of stall, on which the wake separates from the wing.

Experiments showed that the separation of flow from the wing of a aircraft develops very rapidly. If the flow separation is symmetric, then its development is first reflected in a drop of the wing lift, and consequently, in a change of flight path. If the tail unit is located so that the wake from the wing does not fall on it, then the separation of flow on the wing will not influence the deceleration of the flow around the tail unit, but will elicit a decrease of the downwash. A decrease of the downwash will lead to a diving moment, decreasing the angle of attack of the aircraft, for which the aircraft will lower its nose, but have no tendency to stall.

However, in most cases the development of flow separation is asymmetric; flow breaks off nonuniformly, embracing on one wing a large, on the other a smaller part of the area. There are also cases of one-sided flow separation from only one half of a wing.

Asymmetry of the development of a zone of flow separation from a wing has two causes:

- 1) asymmetry of geometric outlines of the wing;
- 2) asymmetry of flow around the aircraft induced by side slipping.

If the flow separation spreads along the wing slowly, then later the formation of a zone of the beginning of a separation on one half of the wing cannot cause sharp dissymmetry in the development of the separation, since on that half of the wing on which the separation started earlier, it will not be able to seize a large area before a separation focus will be formed on the other half of the wing.

If a separation develops very rapidly, before a separation zone forms on the other half of the wing, the lift of the wing with the stripped half flow-around will fall and a moment appears, rotating the aircraft around the longitudinal axis ( $M_x$ ) in the direction of the wing for which coefficient  $c_y$  is reduced. Moment  $M_x$  is so big that ailerons cannot stop the banking of the aircraft.

Figure 77 shows that when transverse equilibrium is disturbed, the angle of stall of a dropping wing ( $\alpha_{\text{stall}}$ ) increases and becomes greater than critical, as a result of which the aerodynamic coefficient of lift ( $c_y$ ) decreases, and the aerodynamic drag coefficient ( $c_x$ ) increases.

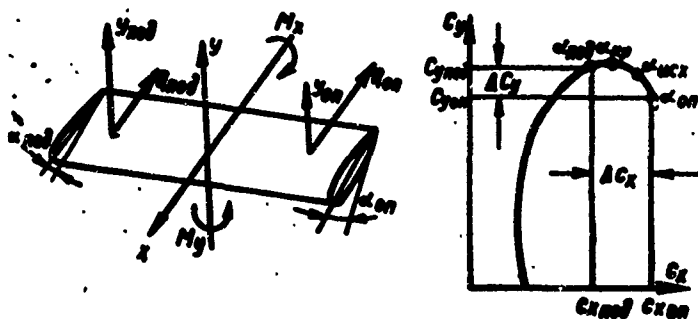


Fig. 77. Aerodynamic forces acting on a wing during spin; coefficients  $c_y$  and  $c_x$  of falling and rising half-cells of wings during a bank.

On a rising wing coefficients  $c_y$  and  $c_x$  decrease. However, the decrease of  $c_y$  is smaller than on the dropping wing, which is conditioned by the more favorable flow around it at an angle of attack less than critical. The difference of lifts creates a moment which rotates the aircraft relatively to the longitudinal axis  $M_x$  in the direction of bank, and the moment of difference of drags rotates the aircraft around the vertical axis in the direction of the dropping wing.

Autorotation of a wing increases from end due to the flow separation at the dropping wing and the side-slipping caused by the earlier separation of flow on the lagging wing.

Spin is classed:

- a) by kind - normal and inverted;
- b) by angle of inclination of longitudinal axis to horizon - steep and flat;

c) by the character of the rotation - stable and unstable.

Inverted spin differs from normal by the fact that the wing of the aircraft is in a region of negative critical angles of stall which can happen if during a spin the control wheel is tilted and the rudder is moved. Since transport aircraft do not engage in aerobatics an inverted spin is not characteristic for them.

A steep spin is characterized by a slope of the longitudinal axis to the horizon of  $50-60^\circ$ , a flat by up to  $40^\circ$ . The fundamental distinction between a steep and flat spin is not the angle of inclination of the longitudinal axis to the horizon, but the rotation of the aircraft: during a steep spin the inside wing is in a critical angle of attack and the outside wing is in an angle of stall less than critical; in a flat spin the whole wing is in a region of critical angles of attack.

The angular velocity of rotation during a flat spin considerably exceeds that during a steep spin, and the vertical velocity of descent during a steep spin exceeds that during a flat spin.

Speed on a trajectory during a steep spin exceeds that during a flat spin. During a flat spin speed is close to the minimum theoretical, and during a steep spin it is the economic or even most advantageous speed.

Stable spin is characterized by the absence of the tendency for the aircraft to change its direction of rotation. Unstable spin is characteristic for aircraft with a sweptback wing, for which a change of the direction of rotation occurs due to the different effective aspect ratio of half-wings during rotation of the aircraft.

Spin is affected by:

1) cargo spacing along axes, which during rotation of the aircraft create moments of centrifugal forces. In Fig. 78 it is

shown that the centrifugal forces of cargo  $F_c$  and  $F_{cl}$ , spaced on the longitudinal axis, create a stalling moment  $M_{stsp}$ , which favors a flat spin;

2) positioning of the center of gravity of the aircraft in the front it delays going into a spin, but in the spin it tends to increase the steepness; in the rear it promotes a spin and favors the transition from steep to flat spin;

3) work of propulsion system and position of flaps:

a) with a working propulsion system with lower decentralization of the propeller, aircraft has a tendency to go into a flat spin, with upper decentralization into a steep spin;

b) deflected flaps promote an earlier spin and the transition into a flat spin.



Fig. 78. Effect on spin of cargo spacing along longitudinal axis.

## 2. Recovery from Spin

In order to recover from a spin, it is first necessary to cease the rotation of the aircraft around the longitudinal and vertical axes. Using ailerons, the control wheel should be tilted in the direction of rotation of the aircraft.

To stop rotating it is necessary to put the aircraft on small angles of stall, and to stop rotation of the aircraft around the longitudinal axis it is necessary to stop its rotation around the vertical axis by tilting the control wheel against spin, moving it completely and with gusto. However, the control wheel tilted, the angles of stall will stay close to or greater than critical and the



aircraft will go into a spin in the opposite direction. Therefore after the rudder is deflected it is necessary to move the control column forward. The degree depends on the type of aircraft; for most planes it is tilted past neutral.

After the rotation stops the pedal is set in neutral and the minimum necessary speed to recover from the steep glide is achieved, the aircraft is put into horizontal flight.

Upon recovery one should strictly observe this sequence of actions:

- 1) if the aircraft goes into a spin with the engine working and flaps deflected, to remove the boost pressure and retract the flaps;

- 2) shove the rudder all the way against the spin and, after the rudder, move the control column back past neutral; in the reverse sequence of moving the rudders, the rudder is blanketed by the elevator, its effectiveness drops, and the aircraft does not come out of the spin as quickly.

### 3. Stalling Characteristics of the An-2

Flight tests of the An-2 established that:

1. When an aircraft descends idling with undeflected flaps and the slats closed, when the control column is smoothly pushed forward, the possibility of recovery on the critical angle of attack is limited by the elevator deflection, since the cross pipe of the steering installation reaches back stop. The aircraft stably pancakes at 105-110 km/h.

2. In flight on idling with flaps deflected  $40^\circ$  and leading-edge slats compressed, smooth and complete deflection of the control column forward at 85 km/h, the ailerons and rudders vibrate. An aircraft with an angle of  $10-12^\circ$  at a vertical descent velocity of

up to 6 m/s pancakes, periodically lifting and dropping its nose, changing with this its forward velocity in the range of 80-95 km/h.

3. Flying at rated power with undeflected flaps and closed slats, smooth deflection of the control column forward to the far position, the aircraft decreases its speed to 55-60 km/h. Further deceleration is limited by the elevator deflection.

Under the influence of the reaction of the propeller the aircraft banks to the left. This requires energetic deflection of control and ailerons.

4. Flying at rated power with flaps deflected 30-40° and closed slats, up to 55 km/h the aircraft stably pancakes. At further deceleration the rudders and ailerons shake violently; keeping its transverse equilibrium, the aircraft pitches down on its nose at 40-45 km/h.

With no interference from the pilot the aircraft after pitching down rapidly gathers speed and at 110-120 km/h comes out of a steep glide. To prevent a great loss of altitude and repeated pitching down of the aircraft when coming out of a glide, it is necessary:

- 1) to remove boost pressure and retract the flaps after the aircraft goes onto its nose,

- 2) after the aircraft reaches the minimum speed (100 km/h) necessary for safe guidance, to deflect the control column forward and put the aircraft into horizontal flight;

- 3) to increase power after recovery from gliding so that as the aircraft approaches the horizon a boost pressure ensuring horizontal flight at 180-190 km/h fixed (boost pressure 650-700 mm Hg, 1600 rpm).

From the instant of pitching to the return to level flight the aircraft loses 200 m altitude.

Thus, it can be concluded that the An-2 has no tendency to stall in a spin upon symmetric flow separation from the wing on critical angles of attack. This is because due to the right-angled wing flow separation starts not at the end tip removable part of the wing, but at the root, and also because of the thick profile and high location of the upper wing, which promote a symmetric development of the flow separation zone, and exclude a sharp decrease of the lift on the lowered wing at an insignificant disturbance of the transverse equilibrium of the aircraft.

Antispin properties of the aircraft are considerably assisted by the automatic slats, which increase the range of flight angles of attack. However, there have been cases of flying on rated power when at a sharp deceleration to 40-45 km/h with deflected flaps or with deflection of flaps at 50-55 km/h the aircraft in the presence of side-slipping or disturbance of the geometric symmetry due to ignorant technical exploitation went into a steep spin.

Note. In all given cases the speed shown is instrument speed.

It is necessary to consider that on the An-2 the instrument and calibrated airspeed (instrument speed taking into account aerodynamic instrument corrections) essentially differ.

To prevent an aircraft from going into a spin:

- 1) carryout recommendations set forth during analysis of the longitudinal controllability of the aircraft (Ch. V, Section 11);
- 2) maintain strict flight conditions as to speed and altitude, allow no disturbance of the coordination of deflection of rudders; always remember that side-slipping especially deep, can lead to pitching with a decrease of speed (increase of angle of attack); side-slipping on the section of prelanding descent, when the aircraft is on large angles of attack, is especially impermissible;
- 3) do not allow sharp deceleration; the elevator deflection for a change of flight conditions must be done by double motions,

calculated so that deceleration does not exceed half the desirable value (the second half of the value of deceleration occurs after the control column is deflected forward;

4) restore assigned conditions in the case of sharp deceleration by increasing the power of the engine and decreasing the angle of attack;

5) thoroughly check the correctness of maintenance and quality of aircraft repair, turning special attention to the absence of waviness on the front section of the upper skin of the wing, synchronous work of the slats, observance of assigned clearances between trailing edge of slats and nose of wing, and also observance of the symmetry of fillets and fairings in where the wing connects with the fuselage and the wing contours.

## C H A P T E R VII

### LEVEL FLIGHT OF AIRCRAFT

#### 1. Forces Acting on an Aircraft in Level Flight, and Conditions of Level Flight

The set level flight is the linear motion of aircraft in a horizontal plane with constant speed. Forces acting on aircraft in level flight are shown in Fig. 79. Level flight requires that:

- 1)  $Y = G$ , which ensures that the aircraft flies in a horizontal plane;
- 2)  $P = Q$ , which ensures that the aircraft flies at constant speed.

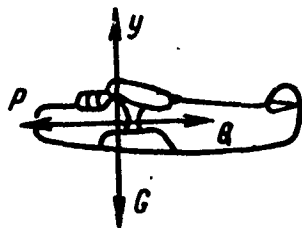


Fig. 79. Forces acting on an aircraft in level flight.

#### 2. Speed Necessary for Level Flight

The speed necessary to develop a lift equal to the weight of the aircraft during a flight on a given angle of attack and a given altitude of flight is called the necessary speed for level flight:

$$V_{r.n} = \sqrt{\frac{2G}{c_y \rho S}}$$

The necessary speed for level flight of a given aircraft depends on the gross weight, flight altitude and angle of attack, but with constant gross weight and flight altitude only on the angle of attack. With an increase of gross weight and flight altitude the necessary speed increases:

$$V_1 = V \sqrt{\frac{G_1}{G}}; \quad V_H = V_0 \sqrt{\frac{P_0}{P_H}}.$$

The quantity  $\sqrt{\frac{P_0}{P_H}}$  is the high-altitude coefficient. With an increase of the angle of attack the speed necessary for level flight decreases, since the coefficient  $c_y$  increases. Every angle of attack corresponds to a definite speed of level flight.

### 3. Thrust Necessary for Level Flight

The thrust necessary to balance drag during flight on a given angle of attack is called the necessary thrust for level flight:

$$P_{r,n} = \frac{G}{K}.$$

An increase of the gross weight of an aircraft increases the necessary speed, causes an increase of drag, which requires augmentation of thrust to overcome.

With an increase of the lift-drag ratio of an aircraft the necessary thrust decreases. Inasmuch as the characteristics of an aircraft with an increase of angle of attack increase to the most advantageous, and then decrease with a further increase of the angle of attack, the necessary thrust with an increase of the angle of attack decreases to the most advantageous after the most advantageous angle of attack increases.

Since every angle of attack corresponds to a definite flight speed a dependence between necessary thrust and flight speed can be established. The graph showing necessary thrust as a function of flight speed is called a Zhukov curve for thrust. Figure 80 is

a Zhukov curve for thrust, which shows that with an increase of flight speed necessary thrust decreases to the most advantageous, and increases with a further increase of flight speed.

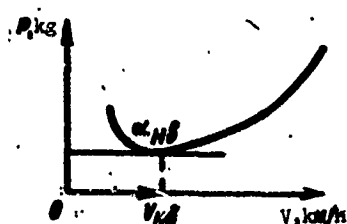


Fig. 80. Zhukov curve for thrust.

Necessary thrust is independent of flight altitude, since with a change of flight altitude impact pressure remains constant and at the same angle of attack lift and drag will remain constant.

#### 4. Power Necessary for Level Flight

The power necessary to ensure level flight on a given angle of attack and given flight altitude is called the required power for level flight,

$$N_{r.n} = \frac{P_{r.n} V_{r.n}}{75}$$

By substituting in this formula the value of necessary thrust and necessary speed for level flight, the expanded formula for the power necessary for horizontal flight can be obtained:

$$N_{r.n} = \frac{V_{r.n}^3}{75 K \sqrt{c_y} \sqrt{S}}$$

From the formula we see that:

1) at a constant angle of attack an increase of the gross weight of an aircraft and flight altitude increases power necessary for level flight; inasmuch as necessary thrust is independent of altitude, and necessary speed increases to the value of the high-altitude coefficient with an increase of flight altitude, then the

required power with an increase of flight altitude will increase to the value of the high-altitude coefficient. Power on an altitude is determined by the formula

$$N_H = N_0 \sqrt{\frac{\rho_0}{\rho_H}};$$

2) at constant gross weight and constant flight altitude the power necessary for level flight depends only on the quality and coefficient of lift and will have a minimum when  $K\sqrt{c_y}$  is maximum.

Inasmuch as  $c_y$  reaches maximum on the critical angle of attack, and the quality of maximum on the most advantageous angle of attack, the maximum value of  $K\sqrt{c_y}$  will be reached on an angle of attack between the most advantageous and critical.

This angle of attack is called the economic and on the An-2 is  $9^\circ$ . Since every angle of attack corresponds to a definite flight speed, then with an increase of speed to the economical rating, necessary for level flight, it will decrease, and at a further increase of speed it will increase. The graph of required power of horizontal flight as a function of speed is called the Zhukov curve for power (Fig. 81).

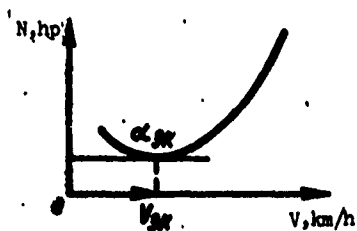


Fig. 81. Zhukov curve for power.

##### 5. Curves of Available and Necessary Powers and Their Use in Determining Flight-Performance Aircraft Data

Graphs fixing the dependence of available and required power on flight speed, represented in one system of coordinates, are called curves of available and required powers (Fig. 82). By comparing



curves of available and required powers  $N$ . Ye. Zhukovskiy determined characteristic speeds of horizontal flight, and so this method obtained name of the Zhukov method. By the Zhukov method one can determine:

1. Critical angle of attack ( $\alpha_{cr}$ ) and corresponding minimum theoretical speed of level flight ( $V_{min}$ ) are found by constructing a perpendicular to the axis of speeds at the tangent to the Zhukov curve. The minimum level flight speed is the least speed which creates a lift equal to the weight of the aircraft. It is theoretical because it does not ensure safety of flight. This requires a speed 20-25% greater than the minimum theoretical.

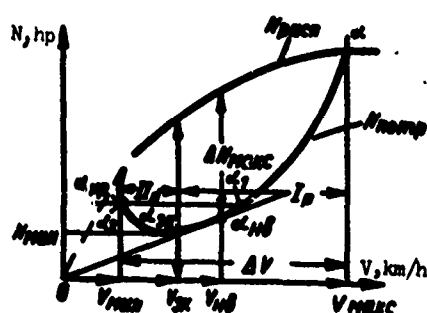


Fig.82. Curves of available and required powers, characteristic speeds of level flight.

2. Economic angle of attack – by constructing the tangent to the Zhukov curve parallel to the axis of speeds. By dropping from the point of contact a perpendicular to the axis of speeds the economic speed can be obtained. It is called economic because during flight on a given speed the consumption per hour of fuel is minimum, which achieves the maximum duration of the flight. The economic speed is the minimum practical speed of level flight.

3. Most advantageous angle of attack ( $\alpha_{ma}$ ) – by constructing the tangent to the Zhukov curve from the origin of coordinates. By dropping a perpendicular to the axis of speeds from the point of contact, the most advantageous speed of level flight  $V_{ma}$  can be found. It is called the most advantageous because during flight at a given speed minimum fuel consumption per kilometer is ensured, which achieves the ultimate flight range.

4. Angle of attack ( $\alpha$ ), corresponding to flight of aircraft at maximum speed – by the intersection of curves of available and required powers; the perpendicular dropped from it is the maximum speed of level flight ( $V_{max}$ ).

5. Speed range ( $\Delta V$ ) – the difference between maximum and minimum speeds; speed range characterizes the degree of maneuverability of aircraft in level flight.

6. Surplus of power ( $\Delta N$ ) – difference between available and required powers on a given flight speed. Maximum surplus of power ( $\Delta N_{max}$ ) corresponds to flight of an aircraft at the most advantageous speed.

7. Two angles of attack, which require identical power: one of them exceeds the economic, the other less than the economic ( $\alpha_1, \alpha_2$ ). Angles of attack less than economic, characterized by high flight speeds, best stability and controllability, compose the first level flight ( $I_p$ ), and angles of attack greater than economic the second level flight ( $II_p$ ). In flying practice only angles of attack of the first regime are used.

The boundary of the two regimes is economic speed. For safety there are no flights at economic speed. Minimum flight speed should exceed the economic by 20-30%. The difference between actual speed of flight and economic is called the speed margin. The less the flight altitude, the bigger the level flight angle of attack speed margin should be. Inasmuch as the An-2 as a rule flies at low altitudes, its minimum practical speed is 140 km/h ( $\Delta N = 20$  km/h).

The two conditions of level flight are determined by conducting secants through the Zhukov curve parallel to the axis of speeds (see Fig. 82).

The An-2 transport with a gross weight of 5250 kg and propeller V-509-D-9A flying near the ground under conditions of an international

standard atmosphere with nonoperating mechanization the high-lift device in the nonworking position, has the following flight performance characteristics:

Minimum theoretical speed.....	96 km/h
Economic speed.....	120 km/h
Most advantageous speed.....	147 km/h
Maximum speed.....	246 km/h
Maximum power surplus.....	237 hp
Speed range.....	150 km/h

With the same conditions for aircraft with an AV-2 propeller maximum flight speed increases by 10 km/h.

6. Influence of Flight Weight, Flight Altitude  
and Agricultural Equipment on the Flight  
Performance Characteristics of Aircraft

Effect of Gross Weight

A decrease of gross weight by decreasing the required power improves the flight performance characteristics of aircraft.

Figure 83 are curves of available and required powers of transport aircraft variant with a gross weight of 4740 and 5250 kg, which show that for aircraft with less gross weight the necessary speeds on all characteristic angles of attack decreased by 5 km/h, and the maximum flight speed increased also by 5 km/h. Thus, a decrease of gross weight for every 500 kg increases the speed range by 10 km/h.

Altitude Effect of Flight

With an increase of flight altitude the required power increases due to the increase of necessary speed (by the value of the high-altitude coefficient). The Zhukov curve when an aircraft is at an altitude will be shifted upwards and to the right relative to the same Zhukov curve when the aircraft flies near the ground. Simultaneously with an increase of flight altitude to the rated level

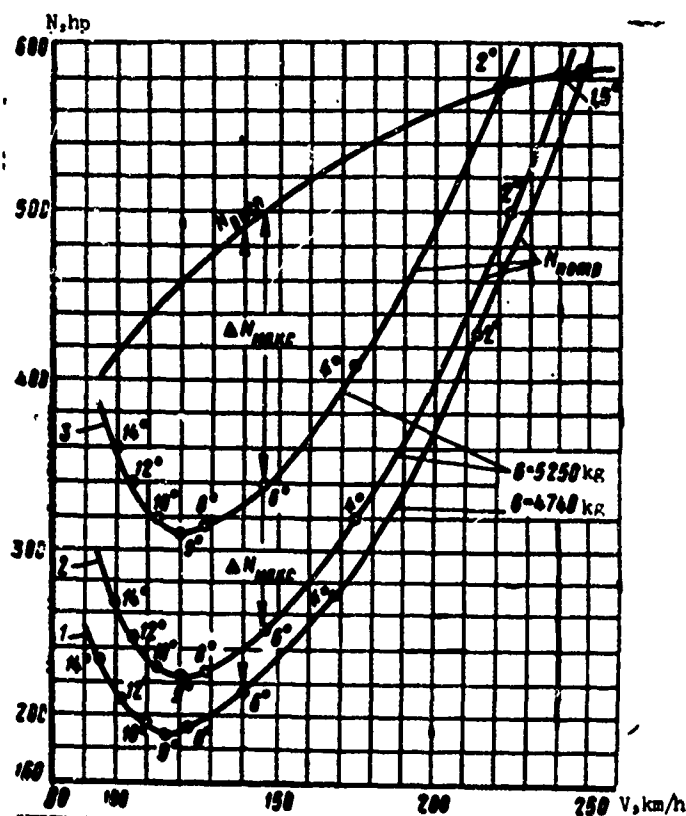


Fig. 83. Effect of gross weight and agricultural equipment on flight-performance characteristics of aircraft with a V-509-D-9A propeller flying at rated power with undeflected flaps. Zhukov curves for different kinds of aircraft: 1 - transport with polar weight of 4740 kg; 2 - transport with polar weight of 5250 kg; 3 - agricultural with gross weight of 5250 kg.

the available power increases, while by a greater value than the required power, as a result of which the flight performance characteristics of the aircraft at the rated altitude are improved - speed range increases by 1 km/h.

When an aircraft flies at altitudes higher than the rated due to the decrease of effective power of the engine the available power decreases, and the necessary increases, therefore flight-performance characteristics fall off.

In Fig. 84 are curves of available and required powers during a flight near the ground, at a rated altitude and higher than the rated altitude.

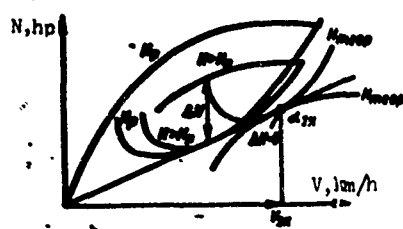


Fig. 84. Altitude effect of flight on flight-performance characteristics of aircraft.

In a flight at the rated altitude with gross weight of 5250 kg an An-2 transport with a V-509-D-9A propeller high-lift device in the nonworking position has:

Minimum speed.....	103 km/h
Economic speed.....	128 km/h
Most advantageous speed.....	157 km/h
Maximum speed.....	254 km/h
Maximum power surplus .....	243 hp
Speed range.....	151 km/h

In a flight under the same conditions, but at 4000 m the aircraft will have the following flight-performance characteristics:

Minimum speed.....	125 km/h
Economic speed.....	145 km/h
Most advantageous speed.....	175 km/h
Maximum speed.....	225 km/h
Maximum power surplus.....	90 hp
Speed range.....	100 km/h

#### Influence of Agricultural Equipment

The flight-performance characteristics of agricultural aircraft due to the increase of required power induced by a decrease of

aerodynamic quality, as compared to transport aircraft fall off considerably. Under identical flight conditions cruising speeds of an agricultural aircraft are 20-25 km/h less than for a transport aircraft, but maximum speed will be 215-220 km/h, i.e., less by 20-25 km/h.

#### 7. Distance and Duration of Flight, Characteristic Cruising Conditions of the An-2

The distance an aircraft can cover with respect to the ground from moment of takeoff to the complete expenditure of the fuel reserve is called the flying range:

$$D = \frac{G_T}{q},$$

where  $G_T$  - fuel reserve, kg;  $q$  - fuel consumption per kilometer, l/km or kg/km.

At a given fuel reserve the flying range depends only on kilometer consumption, which is determined by the formula

$$q = \frac{1}{270} P_n \frac{C_e}{\eta_B},$$

where  $1/270$  is the coefficient obtained by cancelling quantities in deriving the formula;  $P_n$  - necessary thrust, kgf;  $C_e$  - specific fuel consumption per horsepower - hour; depends on rpm, gross weight and altitude of flight;  $\eta_B$  - efficiency of propeller.

From the formula it is clear that the least fuel consumption per kilometer requires:

a) the flight to be at a speed close to the most advantageous, since the necessary thrust will then be minimum;

b) that with respect to flight speed (supercharging) the rpm is selected such that in combination with flight speed the propeller blades are at the most advantageous angle of attack, at which efficiency on given conditions will be maximum. The approximate

most advantageous rpm is determined by the formula  $n_{\text{opt}} = \frac{K_{\text{max}} V_{\text{max}}}{V_{\text{max}}}$  and is definitized during flight tests, the results of which are used to compose a graph.

The conditions of maximum range of flight is used when in an indefinite time with a given reserve of fuel it is necessary to fly the greatest distance. The cruise setting of maximum range of flight of the An-2 transport is characterized by the parameters:

- a) instrument speed ( $V_{\text{np}}$ ) - 145-170 km/h (depending upon gross weight);
- b) boost pressure ( $p_R$ ) - 530-650 mm Hg (depending upon speed of flight);
- c) rpm of crankshaft ( $n$ ) - 1550 per minute;
- d) fuel consumption per kilometer ( $q$ ) - 0.83 l/km.

At a fuel reserve of 500 kg and a full pay load the flying range of the An-2 transport under the given conditions will be 870 km.

The flight of an agricultural aircraft with a gross weight of 5250 kg at a fuel reserve of 500 kg and a speed of 145-150 km/h requires increased operating conditions of the engine:  $p_R = 680$  mm Hg,  $n = 1700$  rpm. In a flight of the aircraft under the given conditions full consumption will increase by 25%, as a result of which the flying range will be only 600 km.

The time which an aircraft can be in the air from the moment of takeoff to full expenditure of the fuel reserve is called the duration of flight

$$T = \frac{G_r}{C_h}.$$

where  $C_h$  - consumption per hour of fuel, l/h, kg/h.

At a given reserve of fuel consumption per hour is determined by the formula

$$C_h = \frac{1}{270} N_{\pi} \frac{C_r}{\eta_p}.$$

where  $N_{\pi}$  - power necessary for level flight.

From the formula it is clear that for the least consumption per hour of fuel:

a) the flight must be at a speed close to economic and at the minimum safe altitude, since the required power will then be minimum;

b) the cruise setting of the biggest duration of flight of the An-2 transport is characterized by parameters:  $V_{np} = 145$  km/h,  $p_K = 530-600$  mm Hg (depending upon gross weight and altitude of flight),  $n = 1500$  rpm, consumption per hour of fuel will be 110-150 l/h.

When in a specific time it is necessary to fly the greatest distance, conditions of the greatest cruising speed, which for the An-2 transport is characterized by the parameters:  $V_{np} = 210$  km/h;  $p_K = 760$  mm Hg,  $n = 1850$  rpm; fuel consumption will be 220 l/h, 1 l/km.



## CHAPTER VIII

### CLIMB OF AIRCRAFT

Steady climb is rectilinear motion of an aircraft on an upwards-slanted trajectory at constant speed.

#### 1. Forces Acting on an Aircraft During Climb, and Steady Climb Conditions

Forces acting on aircraft during climb are shown in Fig. 85:

a) along flight path: force of thrust ( $P$ ), drag ( $Q$ ) and weight component ( $G_2$ );

b) in direction perpendicular to flight path: weight component ( $G_1$ ) and lift ( $Y$ ).

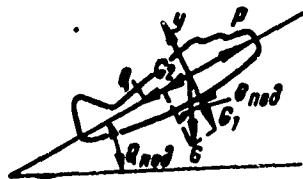


Fig. 85. Diagram of forces acting on aircraft during climb.

Steady climb requires that conditions:

1.  $Y = G_1 = G \cos \theta_{no\alpha}$ , which ensures rectilinearity of the flight path ( $\theta_{no\alpha}$  - climb angle);

2.  $P = Q + G_2 = Q + G \sin \theta_{no\alpha}$ , which ensures constancy of speed.

Thus, during climb the oppositely directed forces are mutually balanced, the aircraft moves by inertia. Lift during climb is less than in horizontal flight at the same angle of attack, since it balances only part of the weight of the aircraft. Thrust necessary during climb is greater than in horizontal flight at the same angle of attack, since, besides drag, it balances the weight component of the aircraft  $G_2$ .

## 2. Speed Necessary for Climb

The speed necessary for climb of an aircraft ( $V_{\text{нол}}$ ) is that speed necessary for creation of lift balancing weight component  $G_1$  during climb of an aircraft at a given angle of attack

$$V_{\text{нол}} = V_{\text{г.н}} \sqrt{\cos \theta_{\text{нол}}}.$$

Since climb on transport aircraft is performed at a low climb angles whose cosine is approximately equal to unity, then the speed for climb is approximately equal to the speed for horizontal flight during climb at the same angle of attack. The speed necessary for climb depends on the same factors as speed for horizontal flight.

## 3. Thrust Necessary for Climb

The thrust needed to balance drag and weight component  $G_2$  during climb of aircraft at a given angle of attack is called the necessary climb thrust:

$$P_{\text{нол}} = P_{\text{г.н}} + \Delta P.$$

From the formula it is clear that climb is possible only at those angles of attack at which in horizontal flight there is a surplus of thrust ( $\Delta P$ ). Inasmuch as during horizontal flight at maximum speed the surplus of thrust is equal to zero, then climb at maximum horizontal flight speed is impossible.

The thrust necessary for climb depends on the same factors as

the thrust necessary for horizontal flight, and also on climb angle, thrust necessary for horizontal flight, and also on climb angle, thrust being proportional to the angle.

#### 4. Climb Angle

The angle between the climb trajectory and the horizon is called climb angle ( $\theta_{\text{max}}$ ) and is determined by formula

$$\sin \theta_{\text{max}} = \frac{\Delta P}{G}.$$

From the formula it is clear that the maximum climb angle for a given gross weight of aircraft can be created during climb of aircraft at an economic angle of attack ( $\alpha_{\text{ex}}$ ), since surplus thrust here ( $\Delta P$ ) is a maximum (Fig. 86).

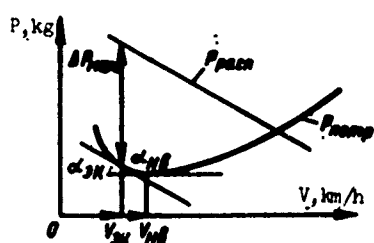


Fig. 86. Surplus of thrust during climb.

Economic rate of climb is the speed of the steepest climb; it is used in flying practice in those cases when it is necessary to surmount an obstacle.

#### 5. Power Necessary for Climb

The power necessary to ensure climb of an aircraft at a given angle of attack, is called required climb power:

$$N_{\text{req}} = N_{r, \text{cl}} + \Delta N.$$

Since climb speed is equal to horizontal flight speed during climb of an aircraft at the same angle of attack, creation of this

speed requires the same power as in horizontal flight. In order to lift the gross weight of an aircraft at a defined vertical velocity, during climb the surplus power available in horizontal flight at a given angle of attack is used. The power necessary for climb depends on the same factors as required power of horizontal flight.

## 6. Vertical Climb Rate

The altitude which an aircraft gains during climb per unit of time is called vertical climb rate:

$$V_{y_{max}} = \frac{75 \Delta N}{G}.$$

Vertical climb rate for a given gross weight of aircraft depends only on the magnitude of surplus power. In order to ensure climb at the highest vertical rate it should be accomplished at the most advantageous speed, since here surplus power is a maximum. The most advantageous rate of climb is called the speed of fastest climb and is used for gaining the assigned flight altitude.

## 7. Polar of Climb Speeds

A graph showing the dependence between climb rate, climb angle and vertical climb rate at different angles of attack is called the polar climb speed (Fig. 87).

From the polar of climb speeds one can determine:

1) angle of attack of maximum horizontal flight speed at which climb is impossible – intersection of the polar with the axis of speeds;

2) best angle of attack ( $\alpha_{best}$ ), at which vertical climb rate ( $V_y$ ) attains a maximum value, by drawing the tangent to the polar parallel to the axis of speeds;

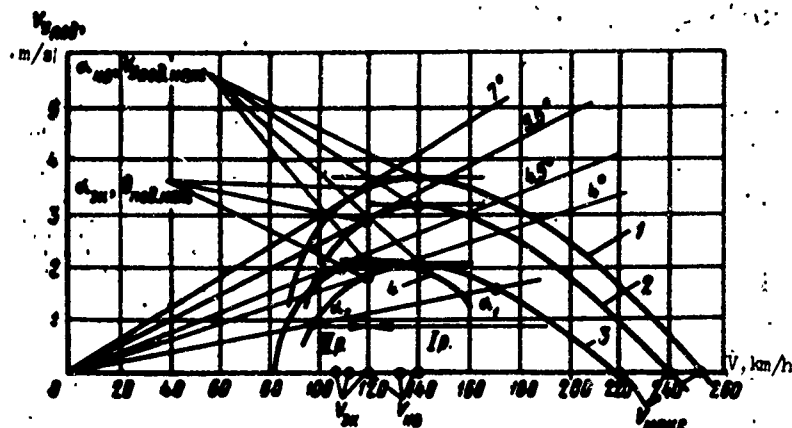


Fig. 87. Polars of climb rates of aircraft during flight near ground with flight weight 5250 kg: 1 - in transport variant with propeller AV-2 without application of flaps; 2 - in transport variant with propeller V-509-D-9A without application of flaps; 3 - in agricultural variant with propeller V-509-D-9A without application of flaps; 4 - in transport variant with propeller V-509-D-9A with flaps set at  $40^\circ$ .

3) economic angle of attack ( $\alpha_m$ ), at which climb angle ( $\theta_{max}$ ) is a maximum, by drawing the tangent to the polar from the coordinate origin;

4) two angles of attack ( $\alpha_1, \alpha_2$ ) at identical climb angle, by drawing the recant from the coordinates origin (see Fig. 87).

An angle of attack which is economic, not having a paired angle, is the boundary between paired angles. Angles of attack less than economic, flight at which is characterized by a higher speed on the climb trajectory, higher vertical climb rate, best stability and controllability of the aircraft, are the first conditions of climb ( $I_p$ ), and angles of attack greater than economic is the second condition of climb ( $II_p$ ). Climb is at angles of attack of the first set of conditions.

## 8. Factors Influencing Climb Characteristics

Aircraft An-2, transport variant with propeller V-509-D-9A, during climb over land with full gross weight and normal rating of

work of the engine without using flaps, has the following data:

a) maximum vertical velocity (at most advantageous speed equal to 140 km/h) is 3.1 m/s;

b) maximum climb angle (economic speed equal to 120 km/h) is 5.5°.

On aircraft with propeller AV-2 the vertical climb rate with other conditions identical is 3.8 m/s, i.e., 0.7 m/s greater.

An essential influence on climb characteristics of aircraft is rendered by gross weight of the aircraft, flight altitude, operating conditions of engine, position of flaps, installation of agricultural equipment, and wind. The altitude effect of gross weight of aircraft and agricultural equipment on climb characteristics are shown in Fig. 88.

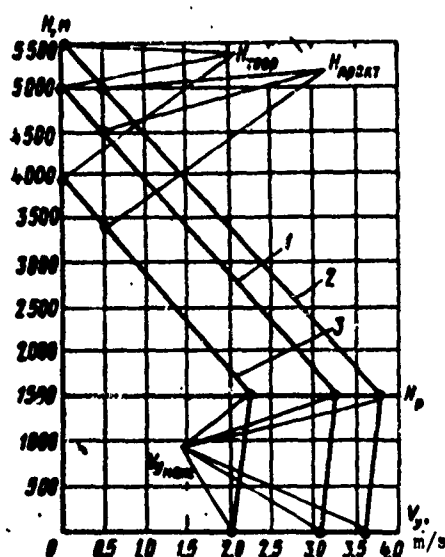


Fig. 88. Flight altitude effect of polar weight of aircraft and agricultural equipment on climb characteristics: 1 - aircraft of transport variant with gross weight 5250 kg during flight with undeflected flaps; 2 - aircraft of transport variant with gross weight 4740 kg during flight with undeflected flaps; 3 - aircraft of agricultural variant with gross weight 5250 kg during flight with undeflected flaps.

#### Influence of Gross Weight

With a decrease in gross weight of aircraft due to a decrease in required power surplus, power is increased, as a result of which the vertical climb rate is increased, and the speed necessary for climb at characteristic angles of attack decreases.

Every 500 kg decrease in gross weight gives an increase in vertical climb rate of 0.6 m/s with a decrease of most advantageous and economic speed by 5 km/h.

#### Flight Altitude Effect

During climb of aircraft with full gross weight to an altitude close to calculation (due to increased surplus power), the vertical climb rate is increased and at calculation altitude will be equal to 3.3 m/s. With further increase in flight altitude vertical climb rate decreases and at a defined altitude will be equal to zero. The altitude at which the vertical climb rate is equal to zero is called the absolute ceiling of the aircraft.

Absolute ceiling of aircraft An-2 transport variant with propeller V-509-D-9A and aircraft gross weight 5250 kg is 5000 m. Since practically the aircraft does not reach its absolute ceiling, due to the fact that the reserve of fuel is depleted earlier than the aircraft will reach this altitude, for every aircraft there is a service ceiling.

The service ceiling of aircraft is the altitude at which the vertical climb rate is equal to 0.5 m/s. The service ceiling of aircraft An-2 with gross weight 5250 kg is 4500 m. Each 500 kg decrease in gross weight increases the absolute and service ceilings of the aircraft 450-500 m.

On aircraft with propeller AV-2 ceilings of aircraft are as a result of an increase in vertical climb rate increased 550-600 m.

#### Influence of Agricultural Equipment

For aircraft of the agricultural variant, due to impairment of the lift-drag ratio, climb characteristics worsen.

Aircraft An-2 agricultural variant with propeller V-509-D-9A with gross weight 5250 kg during climb under normal rating of work of the engine has:

a) maximum vertical climb rate: near ground - 2 m/s; at rated altitude - 2.2 m/s;

b) maximum climb angle -  $4^{\circ}$ .

#### Influence of Operating Conditions of Engine

Operating conditions of the engine render an essential influence on climb characteristics. During climb of aircraft An-2 transport variant with gross weight 5250 kg on takeoff operating conditions of the engine the maximum vertical velocity is 3.8 m/s, and maximum climb angle  $7^{\circ}$ .

Under those same conditions, but with use of conditions of the greatest cruising power (supercharging 760 mm Hg, 1850 revolutions per minute) maximum vertical climb rate will decrease to 1.5 m/s, and maximum climb angle to  $2^{\circ}$ .

#### Influence of Flaps

During deflection of flaps, on account of the decrease in the lift-drag ratio of the aircraft climb characteristics worsen. During climb of the aircraft, transport variant, with gross weight 5250 kg near ground with flaps set at  $40^{\circ}$  the maximum vertical climb at the most advantageous speed equal to 120 km/h will be 2.1 m/s, and maximum climb angle at an economic speed of 110 km/h will be  $4^{\circ}$ .

The decrease in most advantageous and economic speeds is a result of the increase in coefficient  $c_y$  with deflected flaps.

#### Influence of Wind

In horizontal gusts of wind there is a change in ground speed and climb angle. In a tailwind, ground speed is increased, climb angle decreases. In a headwind ground speed decreases, climb angle increases. In a vertical flow all climb characteristics are changed. In an updraft they increase in a downdraft they decrease.



Considering the influence of wing on climb characteristics, gain of assigned altitude in mountain regions should be performed windward.

#### 9. Order of Gaining Assigned Flight Altitude

With an increase in flight altitude to calculated altitude, surplus power is increased, and its maximum value corresponds to the most advantageous angle of attack. Figure 84 shows that with an increase in flight altitude from calculated to ceiling, surplus power ( $\Delta N$ ) decreases, and maximum surplus power is displaced from most advantageous to the economic angle of attack. At ceiling, surplus power is equal to zero, aircraft flight is carried out at economic speed.

Proceeding from this, to ensure the highest rate of climb it should be done at the speed for which surplus power at a given altitude has a maximum value, namely: a) up to rated altitude at a speed equal to 140 km/h; b) after rated altitude the rate of climb is decreased 5 km/h every 1000 m altitude.

Gain of assigned flight altitude as a rule is done at highest cruising power. During climb in mountain regions rated power is used for improvement of stability in controllability of aircraft under conditions of turbulence of the atmosphere; speed is increased 5-10 km/h.

## CHAPTER IX

### DESCENT OF AIRCRAFT

Rectilinear motion of an aircraft on a down-slanted trajectory at constant speed is called steady descent. Descent can be with or without thrust. Descent with zero thrust is called gliding.

#### 1. Forces Acting on Aircraft During Gliding and Condition of Steady Gliding

During gliding the forces on an aircraft are weight of aircraft ( $G$ ) and full aerodynamic force ( $R$ ) — Fig. 89. Inasmuch as motion of the aircraft is on a downwards slanted trajectory, forces act in the following way:

1) the force of weight is broken into two components: in the direction perpendicular to the trajectory of motion,  $G_1$  and in the direction opposite to motion of the aircraft,  $G_2$ ;

2) components of full aerodynamic force;

a) lift balances force  $G_1$  which ensures rectilinearity of motion;

b) drag balances  $G_2$ , which ensures constant speed.

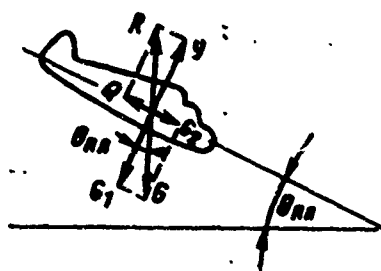


Fig. 89. Diagram of forces, acting on aircraft during gliding.

## 2. Speed Required for Gliding

The speed necessary for creation of lift equal to the component of weight  $G_1$  at a given angle of attack is called the speed necessary for gliding:

$$V_{gl} = V_{cr} \sqrt{\cos \theta_{na}} .$$

Since gliding is done at angles of attack close to the most advantageous for which the gliding angle is insignificant, and aerodynamic coefficients  $c_R$  and  $c_y$  have almost identical values, then the speed necessary for gliding is equal to the speed of horizontal flight of the aircraft during its flight at the same angle of attack.

However, inasmuch as the value of lift during gliding is not influenced by "blowoff" and the vertical component of thrust ( $P_y$ ) as this takes place in horizontal flight at angles of attack close to most advantageous and especially greater than most advantageous, for creation of lift during gliding it is necessary to have a speed somewhat greater than in horizontal flight. In this flight at a  $6^\circ$  angle of attack, coefficient  $c_y$  is equal to 0.74, and during gliding 0.59.

For creation of lift equal to weight of the aircraft during horizontal flight near the ground with propeller V-509-D-94 and gross weight 5250 kg the necessary speed is equal to 147 km/h. When gliding an aircraft with the same gross weight near the ground and at the same angle of attack, the component of weight  $G_1$  is equal to 5105 kg, and lift with preservation of speed of horizontal flight is 4389 kg.

Creation of lift equal to the weight component is possible at a speed of 160 km/h. Proceeding from this, the speed necessary for gliding will be: most advantageous - 160 km/h; economic - 140 km/h.

### 3. Glide Angle

The angle between the glide path and horizon is called by glide angle ( $\theta_{na}$ ). Glide angle is determined by formula

$$\operatorname{tg} \theta_{na} = \frac{1}{K}.$$

Gliding angle depends only on the lift-drag ratio of the aircraft. To ensure the flat test glide it must be done at the most advantageous speed, since here quality is a maximum.

### 4. Vertical Gliding Speed

The altitude which an aircraft loses during gliding per unit time is called vertical gliding velocity ( $V_{y_{na}}$ ). Vertical gliding velocity is determined by formula

$$V_{y_{na}} = \sqrt{\frac{2G}{\rho S}} \cdot \frac{1}{K\sqrt{c_y}}.$$

From the formula it is clear that the maximum vertical velocity of gliding can be attained during gliding of aircraft at an economic speed, since then  $K\sqrt{c_y}$  has a maximum value. Vertical velocity of gliding depends on gross weight of the aircraft and flight altitude, with an increase of which vertical velocity of gliding is increased.

### 5. Gliding Distance

The distance traveled by an aircraft with respect to earth during the time of gliding from a given altitude is called gliding distance ( $L_{na}$ ).

During gliding of aircraft in a calm, glide distance is determined by formula

$$L_{gl} = HK.$$

The maximum range of gliding of aircraft from a given altitude is attained during gliding at the most advantageous speed, and during gliding from an altitude of 1000 m will be 11 km.

## 6. Factors Influencing Descent Characteristics

Descent characteristics of aircraft are influenced by: operating conditions of engine, position of wing mechanization, wing, and gross weight of aircraft.

### Influence of Operating Conditions of Engine

During aircraft descent with thrust of the engine drag decreases (to the value of the thrust force acting in the opposite direction), as a result of which the lift-drag ratio of the aircraft is increased, descent characteristics are improved.

During aircraft descent with thrust (as compared to gliding) there is a decrease in vertical velocity and descent angle, increasing descent distance.

$$V_{y_{ch}} = \sqrt{\frac{2G}{\rho S}} \cdot \frac{Q-P}{Y \sqrt{c_y}}; \quad \text{tg } Q_{ch} = \frac{Q-P}{Y}; \quad L_{na} = H \frac{Y}{Q-P}.$$

Descent on aircraft An-2 with engine thrust is recommended at an instrument speed of 170-175 km/h, with supercharging 500 mm Hg and at 1500 revolutions per minute; vertical descent velocity will be 1.5 m/s, and descent angle 2°. In the given descent conditions the aircraft will in 11 minutes lose 1000 m altitude and travel 32 km in calm, having the most advantageous fuel consumption: 95 l/h, 0.53 l/km.

## Influence of Wing Mechanization

During deflection of flaps, on account of the decrease in lift-drag ratio of the aircraft, descent characteristics worsen: there is an increase in vertical velocity and descent angle decrease in descent distance. Minimum vertical descent velocity at economic speed will be 4.5 m/s, and minimum descent angle  $7.7^\circ$ , ultimate range from an altitude of 1000 m / 5 km.

Joint work of flaps and slats (flaps on leading edge) permits descent in pancaking conditions at an instrument speed of 90-95 km/h, where descent angle is  $12^\circ$ , and vertical descent velocity 5.5 m/s.

An aircraft going into conditions of pancaking is determined by extension of slats, which are advanced: with undeflected flaps at a speed of 105-110 km/h, with deflected flaps at a speed of 90-95 km/h. With a 5 km/h increase in speed as compared to the speed at which slats are extended and an increase of supercharging to over 500 mm Hg the slats are retracted and pancaking is ceased.

## Influence of Wing and Gross Weight

During aircraft descent in a tailwind ground speed and descent distance are increased, and descent angle decreases. During aircraft descent in a headwind, conversely, ground speed and descent distance decrease, and descent angle increases. Descent depends on wind force ( $u$ ) and descent duration of ( $t$ ) from a given altitude:

$$L_{\text{ca, вхвп}} = HK - ut; \quad L_{\text{ca, вхвп}} = HK + ut.$$

During aircraft descent in calm the gross weight influences the necessary speed and vertical descent velocity: the greater the gross weight of the aircraft, the greater are forward and vertical descent velocity.

Besides this, gross weight during descent in a wind influences the distance of gliding, which in a headwind will be even greater

as gross weight of the aircraft increases. The increase in descent distance of aircraft with a large gross weight in a headwind occurs on account of the decrease in duration of descent ( $t_{ca} = \frac{H}{V_{y_{ca}}}$ ):

## 7. Polars of Speeds of Gliding and Powered Descent

A graph showing the dependence of forward velocity and vertical descent velocity at different angles of attack is called a polar of rates of descent. On the polar of rates of descent (Fig. 90) one can determine:

1. Economic angle of attack at which vertical descent velocity ( $V_{y_{cm}}$ ) attains a minimum value, by the tangent to the polar parallel to the axis of speeds. Economic angle of attack, which is equal to  $9^\circ$ , corresponds to an economic speed of 140 km/h.

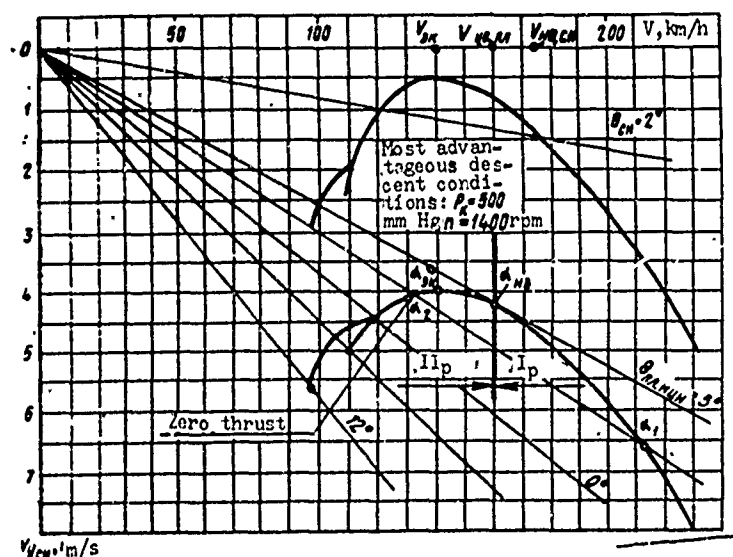


Fig. 90. Polars of speeds of gliding and descent with thrust near ground with gross weight of aircraft 5250 kg.

Minimum vertical velocity of the aircraft ( $V_{y_{min}}$ ) with gross weight 5250 kg in conditions of gliding with undeflected flaps is 4 m/s.

2. Most advantageous angle of attack  $\alpha_{ms}$ , at which descent angle  $\theta_{cr,ms}$  is a minimum, and descent distance a maximum by the tangent to the polar from the coordinate origin. The most advantageous angle of attack is  $6^\circ$ ; it corresponds to a most advantageous speed of 160 km/h.

Maximum gliding angle is  $5.7^\circ$ .

3. Two angles of attack ( $\alpha_1$  and  $\alpha_2$ ) at identical descent angle by the secant from the coordinate origin. The most advantageous angle of attack not having a paired angle is the boundary between angles of attack located on one secant. Angles of attack less than most advantageous make up the first descent conditions ( $I_p$ ), and angles of attack greater than most advantageous the second descent conditions ( $II_p$ ).

Aircraft descent at angles of attack of the second conditions is characterized by lower speeds, the worst controllability and stability. However, for improvement of landing characteristics of the aircraft, in contrast to horizontal flight and climb, angles of attack of the second conditions are used in flying practice. Aircraft descent at angles of attack of the second type lowers flight safety and requires increased pilot attention.

The polar also shows parameters and characteristic of most advantageous descent conditions of aircraft with thrust.



## CHAPTER X

### URNS AND STANDARD TURNS

A correct turn is the motion of an aircraft in a circular arc at constant speed and altitude and without slip. Credit for development of theoretical bases and practical investigation of the turn belongs to the Russian pilot P. N. Nesterov.

#### 1. Forces Acting on an Aircraft During a Turn and Conditions for Execution of a Correct Turn

For creation of the centripetal force acting on an aircraft in the horizontal plane and directed toward the center of the circle, it is necessary to create an angle between the lateral axis of the aircraft and the line of the horizon. This angle is called the angle of bank ( $\gamma$ ).

In formation of the angle of bank (Fig. 91) the vector of lift ( $Y$ ), remaining perpendicular to the lateral axis of the aircraft, is inclined in the direction of bank. The component of lift  $Y_1$  is directed vertically upwards in the plane of action of the force of gravity  $G$ .

The component of lift  $Y_2$  is a centripetal force which is unbalanced and causes the appearance of centripetal acceleration.

For the execution of a correct turn it is necessary that:

- 1)  $Y_1 = G = Y_{\text{BWP}} \cos \gamma$ , which ensures constancy of altitude;
- 2)  $V_2 = Y_{\text{BWP}} \sin \gamma = \text{const}$ , which ensures constancy of radius;
- 3)  $Q = P$ , which ensures constancy of speed.

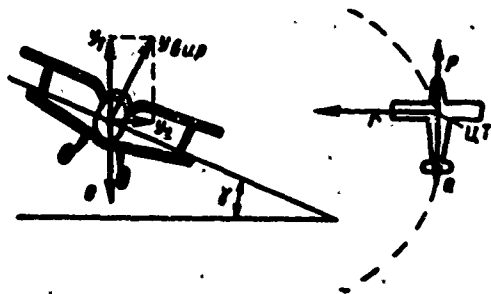


Fig. 91. Diagram of forces acting on an aircraft during a correct turn [ $\text{CT} =$  center of gravity].

From the first equation it follows that for execution of a turn lift must be greater than in horizontal flight for the same angle of attack, and from the first and second equations it is clear that for execution of a correct turn constancy of the angle bank is necessary.

## 2. Overload in a Turn

The lift in horizontal flight balances the flight weight of the aircraft, but in a turn it is greater than with the same angle of attack in horizontal flight. The number showing by how many times the lift in a turn exceeds the weight of an aircraft is called the overload

$$n_{\text{turn}} = \frac{1}{\cos \gamma}.$$

The overload in a turn depends only on the angle of bank, with increase of which it is increased in ever greater degree: for an angle of bank of  $30^\circ$  overload is 1.16, at  $45^\circ$  — 1.43, at  $50^\circ$  — 1.56, at  $60^\circ$  — 2, and at  $70^\circ$  is 2.92.

### 3. Speed Necessary for a Turn

The speed necessary for creation of lift balancing the weight of an aircraft during the execution of a turn at a given angle of attack and creation of centripetal force during a banking turn is called the speed necessary for a turn

$$V_{\text{sup}} = V_{r. n} \sqrt{n},$$

where  $n$  is overload in the turn.

The speed necessary for a turn depends on the same factors as the speed necessary for horizontal flight and on the angle of bank, with increase of which it is increased.

### 4. Thrust Necessary for a Turn

The thrust needed to balance drag during the execution of a turn at a given angle of attack and with a given angle of bank is called the thrust necessary for the turn

$$P_{\text{sup}} = P_{r. n} n.$$

From the formula it is clear that a turn is possible only at three angles of attack, at which in horizontal flight there is a surplus of thrust ( $\Delta P$ ). During the flight of an aircraft at maximum speed a correct turn is impossible, since the surplus of thrust is equal to zero.

### 5. Power Necessary for a Turn

The power necessary for the execution of a correct turn at a given angle of attack, given altitude, and given angle of bank is called the power required for a turn

$$N_{\text{sup}} = N_{r. n} \sqrt{n^3}.$$

From the formula it is clear that the power necessary for a turn depends on the angle of bank, since with its increase

overload ( $n$ ) is increased considerably. In Fig. 92 it is shown that during the execution of a turn with angle of bank of up to  $20^\circ$  the power necessary for the turn is almost equal to the power necessary for horizontal flight; for an angle of bank of  $30^\circ$  the power necessary for a turn is 1.25 times the power necessary for horizontal flight, and for a bank of  $45^\circ$  it is 1.65 greater.

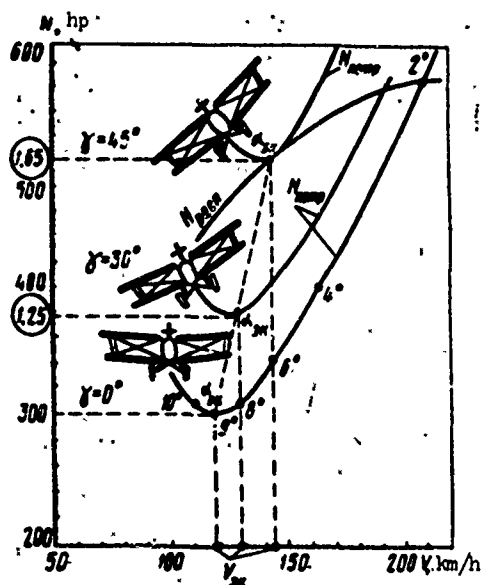


Fig. 92. Dependence of power from the size of the angle of bank during the execution of a turn at economic speed with flight weight of the aircraft of the agricultural version of 5250 kg and V-509-D-9A propeller (in the circles are given the ratios of power necessary for a turn to the power necessary for horizontal flight).

If during a turn bank is progressively increased, there is a moment when all the power of the engine is required for the turn, and further increase of bank causes descent of the aircraft.

Thus, the value of the maximum bank at which a correct turn is possible is controlled by the available power of the propulsion system.

A turn during which all the power of the propulsion system is used for preservation of altitude with maximum bank is called the maximum optimum turn.

## 6. Radius and Time of Turn

The radius and time of a turn are the basic characteristics determining the maneuvering qualities of an aircraft.

The radius of a turn ( $R_{\text{вп}}$ ) and the time of the turn ( $t_{\text{вп}}$ ) are determined by the formulas:

$$R_{\text{вп}} = \frac{V_{\text{вп}}^2}{9.81 \text{tg} \gamma}; \quad t_{\text{вп}} = 0.64 \frac{V_{\text{вп}}}{\text{tg} \gamma}.$$

From the formulas it is clear that for the execution of a turn with minimum radius and in minimum time, minimum necessary speed and the highest possible bank are required.

A turn with maximum bank, with minimum radius, and in minimum time is possible at economic speed, since at this speed the ratio of available engine power to the power required for the turn attains maximum value, and consequently, at this speed it is possible to execute the turn at maximum bank.

During horizontal flight in the transport version of the An-2 aircraft at ground level, with flight weight of 5250 kg, at economic speed, the required power is 223 hp. On the basis of the value of available engine power at economic speed of turn the highest possible bank of the transport version of the An-2 aircraft with flight weight of 5250 kg is  $50^\circ$ .

During the use of the agricultural version of the An-2 with tunnel duster its maximum decreases from 11.3 to 8, as a result of which required engine power for horizontal flight at economic speed, other things being equal, is increased from 223 to 310 hp. In connection with this, the ratio of available power (520 hp) to the power required for a turn decreases to 1.6, and the maximum turn of the agricultural version of the An-2 is possible with a bank of  $43^\circ$ . The economic speed of the turn ( $V_{\text{вп.эк}}$ ) can be determined by the formula

$$V_{\text{вп.эк}} = V_{\text{г.э.}} \sqrt{n}.$$

Inasmuch as the economic speed of the aircraft in horizontal flight at ground level with flight weight of 5250 kg is 120 km/h, the economic speed of a turn under these circumstances will be:

with angle of bank of  $50^\circ$ , 150 km/h; with angle of bank of  $45^\circ$ , 145 km/h; with angle of bank of  $30^\circ$ , 130 km/h.

For flight safety at low altitudes, especially during aerial spraying and dusting, the An-2 is restricted to a maximum permissible bank of  $30^\circ$ . Since the minimum practical speed of horizontal flight of the aircraft is 140 km/h, for execution of maneuvers during aerial spraying and dusting it is 155 km/h.

Dependence of economic speed and minimum practical speed of a turn on the angle of bank is shown in Fig. 93.

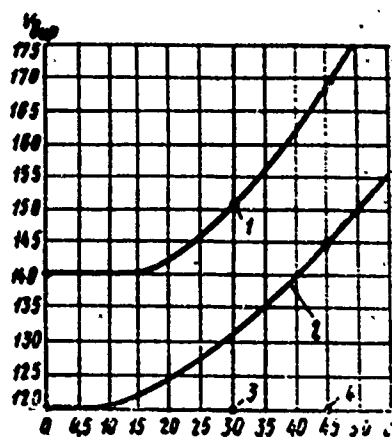


Fig. 93. Dependence of the speed necessary for execution of a turn on the angle of bank: 1 — practical minimum speed of turn; 2 — economic speed of turn; 3 — maximum permissible bank of turn; 4 — highest possible bank of turn.

In the case of threat of collision with obstacles in the air the pilot can maneuver with greater bank, but not exceeding the highest possible angle of bank.

#### 7. The Technique of Executing Turns with the An-2 Aircraft

Before the aircraft is put into a turn the following are necessary:

- a) a reference point must be chosen and the scale of the directional gyrocompass set to 0;
- b) depending upon the angle of bank, the necessary speed must be established;

c) after establishment of necessary speed, the aircraft must be balanced with the trimmer.

In order to turn the aircraft it is banked by turning the steering wheel in the direction of the turn. As the aircraft banks lift decreases, and in order to preserve constancy of altitude, simultaneously with creation of the angle of bank it is necessary to pull back on the steering wheel.

The centripetal force created during the banking of the aircraft causes it to turn around its vertical axis. With the beginning of motion of the aircraft over the circular arc on the half-cells of the wings and lateral surface of the fuselage are created aerodynamic forces, which with respect to the center of gravity create a damping moment, preventing turn of the aircraft about its vertical axis.

The damping moment is created as a result of the difference in the amount of drags on the half-cells of the wings, rotating on different radii, and the lateral aerodynamic force (Z).

In order to surmount the braking moment created by the difference in drag on the half-cells of the wings and the lateral aerodynamic force of fuselage, the entry into turn must be made in a coordinated fashion. As the aircraft banks the rudder is deflected in the direction of the turn, with its effects being noted on the turn-and-bank indicator (the ball of the indicator should be in the center).

Thus, the overall motion of the controls during the entry of the aircraft into a turn should be the following: steering wheel in the direction of the turn, steering column slightly back, and simultaneously with deflection of steering wheel the rudder is put in the direction of the turn.

After the establishment of assigned bank, for reduction of the rolling moment, which is due to the difference in speeds in the

half-cells of the wings, steering wheel must be gradually turned to neutral position (opposite the turn). With decrease of the angle of deflection of the ailerons in the process of execution of the turn, in order that the aircraft does not go through outside slip it is necessary to deflect the rudder in the direction opposite the turn.

Before the selected reference point for the angle of bank is reached, it is necessary to start recovery from the turn. The movement of the controls here should be the following: steering wheel and rudder in the direction opposite the turn, and steering column slightly back.

During the execution of a turn with  $45^\circ$  bank it is necessary to increase boost with increase of the angle of bank in such a manner that it is full at  $45^\circ$ .

During a turn with bank of  $45^\circ$  it is as if the controls are reversed. When it is necessary to accelerate the turning of aircraft, the steering column must be pulled back, increasing the angle of attack. Due to increase of coefficient  $c_y$  the centripetal force is increased. Simultaneously the vertical component of lift is also increased, raising the trajectory of the turn somewhat.

If the pilot has to raise the trajectory of the turn or to delay its descent without accelerating the turning of the aircraft, he must turn the rudder in the direction opposite that of the turn. With deflection of the rudder inside slip is created, as a result of which a lateral aerodynamic force arises, creating the moment which raises the trajectory. Hence it is clear that the controls in the character of the effect remain unchanged: the rudder changes the angle of slip, while the elevator changes the angle of attack.

During the execution of a turn one should also bear in mind that on the An-2 aircraft, as on other aircraft, acts a gyroscopic moment, appearing as a result of change of the plane of rotation of the propeller. Under the action of the gyroscopic moment the aircraft tries to raise or lower its nose.



The direction in which the nose of the aircraft is deflected can be determined according to the existing mnemonic rule.

If the direction of turn of the aircraft coincides with the direction of rotation of the propeller, the aircraft tries to lower its nose, and if the directions are opposite, it tries to lift its nose. Consequently, the An-2 has a tendency in a left turn to lift its nose and to lower it in a right turn.

## 8. Standard Turns

The maneuver of an aircraft along the horizontal for change of direction of flight by  $180^\circ$  with return to the same vertical plane from whence the turn is started is called the standard turn.

The standard turn consists of two parts:

- a) a turn of  $80^\circ$  in the direction opposite the standard turn;
- b) a turn of  $260^\circ$  in the direction of the standard turn.

Since during transition of the aircraft from one turn to the other it turns an additional  $20^\circ$ , the sum of all angles during the execution of the standard turn is  $360^\circ$ .

The standard turn is used for the landing approach in adverse weather conditions and during aerial spraying and dusting. For execution of standard turns the following are required:

- a) during aerial spraying and dusting: speed of not less than 155 km/h, angle of bank of up to  $30^\circ$  (in bumpy air not over  $20^\circ$ );
- b) during a landing approach in adverse weather conditions: speed of 160 km/h, angle of bank of  $15^\circ$ .

The effects of the controls during execution of the standard

turn are analogous to those during execution of the turn. The times required for execution of the standard turn are: with bank of  $30^\circ$ , 55 s; with bank of  $20^\circ$ , 90 s; with bank of  $15^\circ$ , 140 s.

9. The Most Dangerous Errors Committed During  
the Execution of Maneuvers  
During Aerial Spraying

The usual flight of a transport aircraft consists basically of steady rectilinear conditions, for which overload is equal to unity ( $Y = G$ ), and only during the execution of turns does overload exceed unity. During aerial spraying and dusting the aircraft flies with overload greater than unity for a considerable part of the time.

After a relatively brief segment of horizontal flight on an altitude of several meters from the ground, the aircraft is shifted into a climb with reduction of speed to the most advantageous level, and then at an altitude of 50 m it again accelerates to a safe speed and is put into the standard turn for approach to the next "pass."

In addition to the peculiarities of operation connected with maneuvering near the ground, the aircraft of the agricultural version is distinguished by a relatively smaller margin of thrust than the transport version and also a significant change in flight weight in process of each short (measured in minutes) flight as a result of the rapid expenditure of chemicals. These peculiarities of flight during aerial spraying and dusting seriously complicate piloting and require the continuous increased attention of the pilot. The most dangerous errors in piloting are those connected with the tendency of the pilot to decrease turning time by decelerating or increasing overload.

Forcing a Turn by Means of Deceleration<sup>1</sup>

Earlier we examined the turn of an aircraft for which the vertical component of lift  $Y_1$  was equal to the flight weight of

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<sup>1</sup>Basic statements of this and the following of divisions are taken from the work of M. V. Rozenblat "Pilota o peregruzke" ("The Pilot on Overload"), Redizdat Aeroflot, M., 1964.

the aircraft, and overload in process of execution of a turn with given angle of bank was constant. However, it is possible to execute a turn with constant overload and angle of bank with continuously varying angle of attack of the aircraft and, consequently, with varying value of coefficient  $c_y$ . This is feasible only under the condition of such change of the speed of flight that the product  $c_y V^2$  remains constant ensuring constancy of lift.

Consequently, continuous increase, in the process of turning without climb, of the value of coefficient  $c_y$  should be accompanied by the same decrease of the value of the square of speed. Therefore the radius of the turn will continuously decrease, and the time of the turn will be reduced. Turning with varying speed and constant overload brings the aircraft nearer to wing stall.

If the execution of a forced turn with constant angle of bank occurs with continuous deceleration, such deceleration without climb is possible only with decrease of engine thrust as compared to that necessary at the initial speed of the turn. However, decreasing the thrust of the propulsion system increases the danger, since the lift in this case decreases as a result of the weaker airflow and reduction of the vertical component of thrust ( $P_y$ ).

#### Forcing a Turn by Increasing Overload

Increasing overload in a turn with constant angle of bank by increasing the angle of attack of the aircraft leads to simultaneous increase of both components of lift — vertical  $Y_1$  and horizontal  $Y_2$ . The increase of centripetal force ( $Y_2$ ) decreases the radius, while the increase of the  $Y_1$  component of lift leads to upward distortion of trajectory, i.e., to transition of the aircraft from horizontal flight to climb as a result of excess of the  $Y_1$  component of lift as compared to flight weight.

Such piloting inevitably leads to loss of flying speed as a result of expenditure of kinetic energy on climb of the aircraft

and to an increase of its drag at the greater angle of attack. The simultaneous increase of overload as compared to that necessary for turning with given angle of bank and reduction of speed bring the aircraft close to a stall. In this case during flight at low altitude the pilot has very little time for preventing the stall. Here the complexity of the situation is additionally increased by the fact that the increase in engine operating conditions does not create a sufficiently great margin of thrust for fast acceleration of the aircraft, since the aircraft of the agricultural version, equipped external hangers, especially the sprayer, has greater drag.

Increase of overload in a turn can also occur without increase of the  $Y_1$  component of lift, which is possible under conditions of continuous increase of the angle of bank in process of turning. In this case component  $Y_1$ , equal to  $V_{BMP} \cos \gamma$ , with increased value of lift is held constant as a result of decrease of  $\cos \gamma$  at the same rate. Centripetal force  $Y_2$ , equal to  $V_{BMP} \sin \gamma$ , is increased continuously, as a result of which forcing of the turn takes place. And again, as in the first case, there is reduction of flight speed due to the necessity of increasing the angle of attack with increase of the angle of bank, causing the aircraft to approach a stall.

In all cases the forcing of a turn, connected in the first place with continuous deceleration, considerably complicates piloting of aircraft at low altitudes, since under conditions of fast change of conditions of turn the controllability characteristics of the aircraft change simultaneously.

Furthermore, it should be borne in mind that a much greater danger especially at large angles of attack, is presented by outside slip, created by large rudder deflection. Lift in this case on the outer half-cell of the wings increases, as a result of which the angle of bank is sharply increased.

In trying to decrease the angle of bank, the pilot turns the steering wheel in the opposite direction, which aggravates the

situation, since the aileron on the inner half-cell of the wings, being lowered, increases induced drag, as a result of which bank is increased still more. The increase of bank leads to creation of a difference between the weight of the aircraft and the vertical component of lift ( $\Delta G = G - Y_1$ ), causing the aircraft to drop, and the pilot, increasing the angle of attack, puts the aircraft into a stall.

To ensure safety during the execution of turns, it is necessary to maintain strict coordination of deflection of controls and steady conditions of speed and altitude, especially during aerial spraying and dusting, in segments of circling flight from takeoff to the second turn, and from the third turn to the final approach, where low speed and low altitude are combined. Under given flight conditions it is always necessary to consider:

1. That the greater the angle of bank in a turn, the bigger the overload and, consequently, the higher the speed at which a stall can occur ( $V_{cr, sup} = V_{cr, r. n} \sqrt{n}$ ).

Thus, for example, during a turn with bank of  $30^\circ$  stalling occurs at a speed exceeding the stalling speed in horizontal flight by 8%, and with a bank of  $45^\circ$  this increases to 20%.

2. Peculiarities of change of altitude of flight during aerial spraying and dusting: during transition of an aircraft from horizontal flight to descent the vertical velocity of descent is increased, and altitude is continuously reduced.

After the pilot has pulled back on the steering column to increase the angle of attack, the aircraft will continue to drop for a long time. Because of this, the execution of standard turns at altitudes of less than 50 m is impermissible, and transition to climb from low-level flight must be started in sufficient time to prevent drop of the aircraft below an altitude of 5 m (above forested and rugged terrain — 10 m).

3. In case of engine failure in flight it is first necessary to establish proper gliding speed by pushing the steering column forward, and then decide on further action, depending upon the situation.

With engine failure during takeoff, on the straight line to the first turn, turning back to the airfield is not allowed, but the landing must be made straight ahead, with the aircraft being maneuvered to avoid direct collision with obstacles: in a gliding turn of  $90^{\circ}$  an aircraft loses 60-70 m of altitude.

## CHAPTER XI

### AIRCRAFT TAKEOFF

Takeoff is the accelerated motion of an aircraft from the start of the run to a climb of 25 m.

#### 1. Diagram of Takeoff and Forces Acting on the Aircraft During Takeoff

Takeoff consists of four stages: run, lift-off, acceleration (flare-out), and climb to 25 m.

Run is the initial period of takeoff, necessary for reaching the speed at which lift sufficient to raise the aircraft from the ground is created.

The moment of separation of the aircraft from the ground is called by lift-off.

The distance covered by the aircraft from moment of lift-off to achievement of speed ensuring safety of transition of the aircraft to climb is called by acceleration.

If acceleration is accomplished at constant altitude, the take-off is called classical. If acceleration is accomplished with gradual withdrawal of the aircraft from the ground, the takeoff is called normal.

Takeoff of the An-2 aircraft is made according to the normal scheme.

The peculiarity of takeoff of the An-2 consists in the fact that its run, up to lift-off, is made without separation of the tail wheel from the ground, with the control column in the neutral position.

This peculiarity of takeoff is caused by the necessity to reduce takeoff distance by increasing the angle of attack at lift-off.

During a takeoff without breakaway of the tail wheel from the ground during the run, because of the low speed, drag is increased slightly and the margin of thrust is comparatively high.

Increasing the angle of attack at lift-off makes it possible, by increasing coefficient  $c_y$ , to create the lift needed for separation of the aircraft from the ground at lower speed.

In Fig. 94 are shown the forces acting on an aircraft during takeoff:

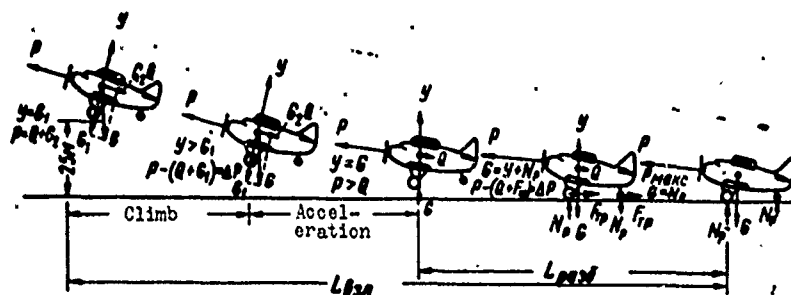


Fig. 94. Diagram of forces acting on an aircraft during takeoff, and stages of takeoff distance [ $L_{B3H}$  - takeoff;  $L_{P30}$  - run].

1) in the direction of takeoff - tractive force ( $P$ ), attaining maximum value at full boost at the beginning of the run, then gradually decreasing with increase of speed;

2) in the direction opposite takeoff;



a) drag ( $Q$ ), which at the beginning of the run is created mainly as a result of air blown over the aircraft by the propeller slipstream, and then increases with increase of speed;

b) frictional force of the wheels ( $F_{Tp}$ ), which decreases with increase of speed in the run and vanishes at the time of lift-off;

c) the component of weight  $G_2$ , appearing with transition of aircraft to climb;

3) in the direction perpendicular to the trajectory, lift ( $Y$ ), which in the beginning is created mainly by the slipstream, then increases with increase of speed;

4) in the direction perpendicular to the plane of the ground:

a) downwards — the weight of the aircraft ( $G$ );

b) upwards — the reaction of the earth ( $N_p$ ).

With the aircraft standing ready for takeoff, its weight is balanced by the reaction of the earth ( $G = N_p$ ). With the beginning of creation of lift the reaction of the earth decreases, and the weight of the aircraft is balanced by the lift and the reaction of the earth ( $G = Y + N_p$ ).

The decrease of reaction of the earth ( $N_p$ ) is accompanied by decrease of frictional force ( $F_{Tp}$ ). At the time of lift-off of the aircraft the lift balances the weight of the aircraft ( $Y = G$ ), and the reaction of the earth and the frictional forces are equal to zero ( $N_p$  and  $F_{Tp} = 0$ ).

Acceleration of the aircraft during the run is due to the unbalanced force  $\Delta P$ , equal to  $P - (Q + F_{Tp})$ . After lift-off  $\Delta P = P - Q$ , and during climb  $\Delta P = P - (Q + G_2)$ .

In process of acceleration the pilot, pushing the wheel forward, maintains equality of lift and weight, with gradual departure of

the aircraft from the ground, gathers necessary speed, and then put the aircraft into a climb.

Transition of the aircraft from acceleration to climb of 25 m constitutes unsteady motion on a trajectory bent gradually upwards with a certain increase of speed. The centripetal force in this segment is  $\Delta Y = Y - G_1$ , where  $G_1$  is component of weight active in the direction opposite lift.

As the trajectory is bent upwards the  $G_2$  component of weight is increased and the  $G_1$  component decreases, as a result of which acceleration of the aircraft decreases.

The pilot, moving the steering column forward, reduces the angle of attack in order to decrease drag and the centripetal force in such a way that at an altitude of 25 m  $Y = G_1$ .

Decreasing engine speed at 25 m, the pilot makes a steady climb.

## 2. Forms of Takeoff and Takeoff Characteristics of Aircraft

Depending upon conditions, upon the decision of the crew commander, takeoff in the An-2 can be accomplished by the following methods:

- a) without application of flaps at rated engine power;
- b) with application of flaps at  $25^\circ$  and at rated engine power;
- c) with application of flaps at  $30^\circ$  and with use of takeoff power.

The takeoff of an aircraft is characterized by lift-off speed, run distance, and length of takeoff distance.

Lift-off speed ( $V_{\text{отр}}$ ) is the speed at which for the established lift-off angle the lift necessary to separate the aircraft from the ground is created. Lift-off speed is determined by the formula

$$V_{\text{отр}} = 3,6 \sqrt{\frac{2\sigma}{C_{y_{\text{отр}}} S}}.$$

The run distance ( $L_{\text{разб}}$ ) is the distance covered by an aircraft from start to the point of lift-off. This distance is determined by the formula

$$L_{\text{разб}} = \frac{(V_{\text{отр}} \pm u)^2}{2j},$$

where  $u$  — wind speed;  $j$  — acceleration of aircraft during the run.

Acceleration can be determined by the formula

$$j = 9,81 \frac{P - (Q + F_{\text{тр}})}{G}.$$

From the formula it is clear that the value of acceleration depends:

- a) on the value of available thrust developed by the propeller during the run;
- b) on the frictional force of wheels;
- c) on drag of the aircraft;
- d) on the flight weight of the aircraft.

The value of frictional force ( $F_{\text{тр}}$ ) depends on flight weight and lift and also on the coefficient of friction ( $f$ ) between the tires and the surface of the airfield, which is clear from formula

$$F_{\text{тр}} = f(G - Y).$$

The coefficient of friction between the tires and the surface of the airfield depends on the character of the airfield surface and for assigned tire pressure ( $2.5-3 \text{ kg/cm}^2$ ) has the values shown in Table 4.

Table 4.

Character of surface	Value of coefficient of friction
Concrete runway.....	0.03
Hard grassy ground...	0.05
Soft grassy ground...	0.08
Soft sand.....	0.10
Ice.....	0.03
Rolled snow.....	0.15
Loose wet snow.....	0.30

With lowered tire pressure the coefficient of friction is increased considerably.

Takeoff distance ( $L_{\text{взл}}$ ) is the distance covered by the aircraft with respect to the ground from the start of the run to climb of 25 m. Takeoff distance is determined by the approximate formula

$$L_{\text{взл}} = (2.5 - 3) L_{\text{под.}}$$

The takeoff characteristics of the An-2 aircraft with V-509-D-9A propeller at flight weight of 5250 kg, in calm air are characterized by the data in Table 5.

Table 5.

Version of aircraft and character of air- field surface	Forms of takeoff					
	at rated power				at takeoff power	
	no flaps		with flaps at 25°		with flaps at 30°	
	Lift-off speed, km/h	run dis- tance, m	Lift-off speed, km/h	run dis- tance, m	Lift-off speed, km/h	run dis- tance, m
<b>T r a n s p o r t</b>						
Takeoff on wheels from hard grassy ground.....	110	$\frac{250}{800}$	90	$\frac{200}{600}$	85	$\frac{170}{540}$
Takeoff on skis on rolled snow.....	110	$\frac{280}{830}$	90	$\frac{220}{620}$	85	$\frac{180}{550}$
Takeoff on skis on unrolled snow...	110	$\frac{300}{850}$	90	$\frac{240}{630}$	85	$\frac{200}{580}$
<b>A g r i c u l t u r a l</b>						
Takeoff on wheels from hard grassy ground .....	110	$\frac{280}{860}$	90	$\frac{240}{700}$	85	$\frac{190}{630}$

Notes: 1. Data are given for conditions of international standard atmosphere.

2. Run distance on skis is given for ambient air temperature of  $-10^{\circ}\text{C}$ .

### 3. Factors Influencing the Length of Run of an Aircraft

Analysis of formulas for run distance, lift-off speed, acceleration of aircraft, and frictional force of wheels against the air-field surface makes it possible to conclude that the run distance of an aircraft for takeoff at assigned lift-off angle ( $\alpha_{\text{OTP}} = 14^{\circ}$ ) is influenced by the following factors.

Engine operating conditions. Application of takeoff operating conditions of engine as compared to nominal increases the margin of thrust ( $\Delta P$ ), as a result of which acceleration of the aircraft during the run is increased, and lift is increased as a result of the stronger slipstream (with the V-509-D-9A propeller at the time

of lift-off the margin of thrust is: under rated conditions 320 kg, under takeoff conditions 450 kg). The increase of the margin of thrust of approximately 130 kg and increase of lift during takeoff under takeoff operating conditions of the engine reduce the length of the run by 30-40 m.

Application of flaps. Deflection of flaps on takeoff, in view of the low speed during the run, leads to a slight increase of coefficient  $c_x$ , which with considerable margin of thrust ( $\Delta P$ ) does not significantly influence the decrease of acceleration during the run. At the same time a considerable increase of  $c_y$  makes it possible to create the lift necessary for lift-off at lower speed and to decrease the frictional force of the wheels against the surface of the airfield ( $F_{Tp}$ ), as a result of which the run distance decreases. During the takeoff of an aircraft with flaps at  $25^\circ$ , as compared to takeoff with undeflected flaps, the run distance is reduced by 40-50 m.

Flight weight of aircraft. Decreasing the flight weight of an aircraft decreases the speed necessary for lift-off from the runway and the frictional force of the wheels against the airfield surface, as a result of which the run distance decreases.

For each 500 kg decrease of flight weight run distance is decreased by 50 m.

Direction and force of wind. An increase of wind force due to change of local velocity of flow over the wing cells reduces run distance with headwind and increases it with tailwind. For each 5 m/s increase of the force of headwind run distance decreases by 50-60 m.

Altitude of airfield above sea level and temperature of ambient air. With increase in altitude of the airfield above sea level and increase of temperature of the ambient air, air density ( $\rho$ ) decreases, which leads to increase of the speed necessary for lift-off and decrease of available propeller thrust, as a result

of which run distance is increased. For each 500-m increase in elevation of the airfield above sea level or increase of ambient air temperature of  $10^{\circ}$  (above  $+15^{\circ}\text{C}$ ) run distance increases by 7%.

Surface of airfield. The character of the surface of the airfield determines the value of frictional force and thereby the actual run distance. During the takeoff of an aircraft under takeoff power or with flaps at  $30^{\circ}$  from an airfield with soft or sandy ground the run distance as compared to that from hard ground, other things being equal, is increased by 60-70 m.

Incline of airfield. During takeoff on an incline the acceleration during the run decreases as a result of the effect of weight component ( $G_2$ ), and in connection with this the run distance is increased. During takeoff on an incline of 0.01 (elevation of airfield increases 1 m for every 100 m from the start of the runway (VVP)) the run distance is increased by 20-25 m.

#### 4. Determining Length of Run

The run distance of the An-2, with allowance for factors affecting it, is determined on nomographs developed by the State Scientific Research Institute of Civil Aviation (Fig. 15).

For determination of run distance on the nomograph the following are necessary:

a) in accordance with true temperature (lower scale) and atmospheric pressure to find on graph "a" the point of intersection of the line drawn from the lower scale (vertically upwards) from one of the curves characterizing true pressure;

b) the found point of intersection is transferred (horizontally to the right) to graph "b" to the intersection with the line indicating true wind speed;

c) from graph "b" the point found is transferred (vertically

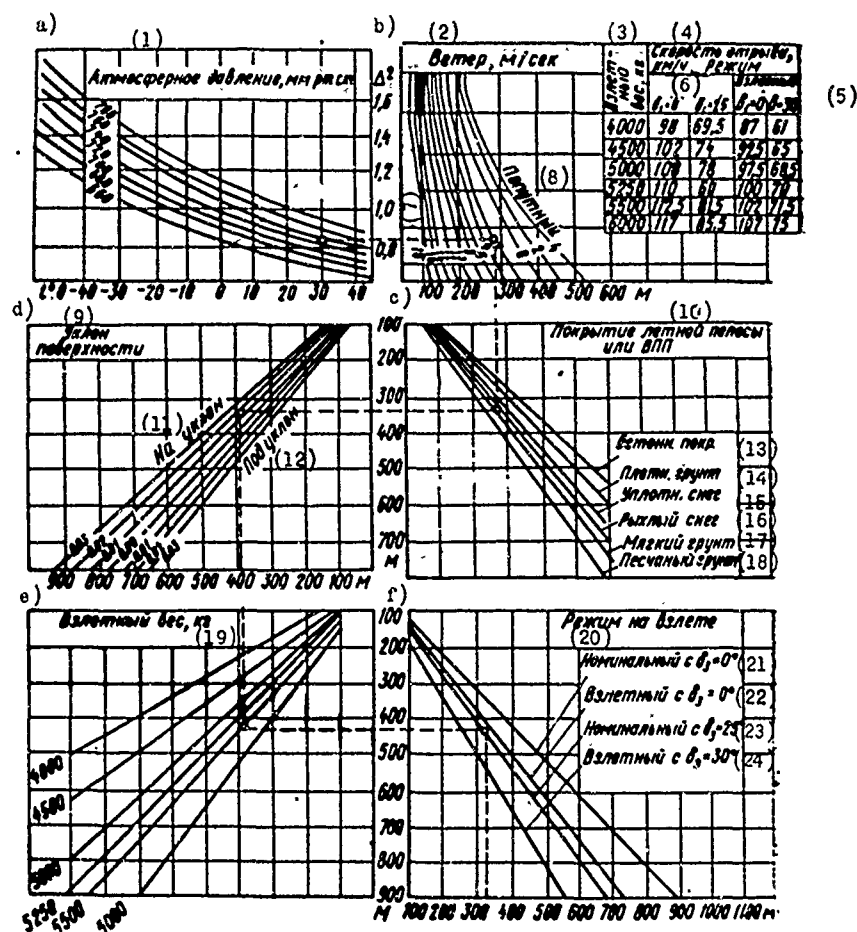


Fig. 95. Nomographs for determination of run distance for lift-off angle of  $13^\circ$ .

KEY: (1) Atmospheric pressure, mm Hg; (2) Wind, m/s; (3) Takeoff weight, kg; (4) Lift-off speed, km/h power; (5) Takeoff; (6) Rated; (7) Headwind; (8) Tailwind; (9) Surface slope; (10) Covering of airfield or runway; (11) Up slope; (12) Down slope; (13) Concrete; (14) Solid ground; (15) Packed snow; (16) Loose snow; (17) Soft ground; (18) Sandy ground; (19) Takeoff weight, kg; (20) Conditions on takeoff; (21) Nominal; (22) Takeoff; (23) Nominal; (24) Takeoff.

downwards) to graph "c" to the intersection with the line characterizing the state or type of covering;

d) from graph "c" the point of intersection is transferred (horizontally to the left) to graph "d" to the intersection with line of slope (up or down slope);



e) from graph "d" the point of intersection is transferred (vertically downwards) to graph "e" to the intersection with the line of takeoff weight of the aircraft;

f) from graph "e" the point of intersection is transferred (horizontally to the right) to graph "f" to the intersection with the straight line indicating conditions and angle of deflection of flaps; after the obtained point is transferred (vertically downwards) the run distance corresponding to actual conditions of takeoff is read on the lower scale.

## 5. Technique of Taking Off Without Flaps

The takeoff without flaps is the basic form of takeoff for the An-2 and is permitted with headwind of not over 18 m/s.

For execution of takeoff it is necessary:

1. To occupy the initial position for takeoff, placing the aircraft parallel to the axis of the runway and taxiing in a straight line for 5-10 m so that the tail wheel is set on the longitudinal axis of the aircraft. Pick a reference point on the horizon for maintaining direction.

2. After checking readiness for takeoff according to the check list, request takeoff clearance; having obtained it and smoothly increasing engine power to nominal, with column in neutral position, start run.

3. Direction of run to selected reference point is maintained in the first half by using the brake, in second by means of the rudder. During the run the aircraft has tendency to turn to the left under the influence of turning moments:

- a) reaction of the propeller, which during clockwise rotation increases the reaction on the left wheel, increasing the frictional force of this wheel (for constant engine power and propeller speed

reaction remains constant); the value of propeller reaction at the start of the run depends on the rate of increase of engine power: a sharp increase in throttle is accompanied by a sharp increase of propeller reaction and makes maintenance of the direction of the run difficult;

b) gyroscopic moment of propeller, which acts in that case when the pilot moves the steering column during the run, lifting the tail wheel from the runway.

The action of the gyroscopic moment is stronger, the more sharply the tail wheel is lifted.

Since the speed of the aircraft at the beginning of the run is low, the vertical empenage is of little effect, and turning moments in this period attain maximum value, in order to reduce them one should not allow a sharp increase of engine power or deflection of the column forward from neutral position, and for countering turning moments it is necessary to apply the brake.

During execution of the run one should always remember that if a turn is made, it will be amplified by the effect of lateral forces of friction, and therefore with sharp deflection of the aircraft the takeoff has to be aborted.

If an attempt is made to continue the takeoff, the large force of inertia ( $F_{\text{И}}$ ) and strong lateral friction ( $F_{\text{TP}}$ ) with respect to the point of contact with the ground by the outer wheel will create a moment which can exceed the moment of force of the weight of the aircraft relative to this point, and the aircraft will pitch over on the outside wing (Fig. 96).

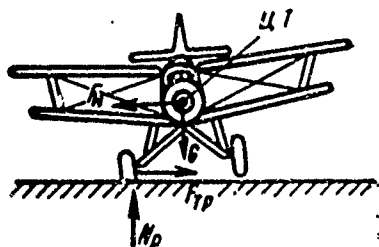


Fig. 96. Forces acting on the aircraft in a turn during takeoff run.

4. With increase of speed during run the aircraft has a tendency to lift its tail, especially during takeoff without load, in view of which it is necessary to hold the column back, preventing lift-off at low speed and wheel bounce.

The tendency of the aircraft to pitch after separation from the runway should be countered by pushing the column forward. It must be borne in mind here that for balancing the aircraft slight downward elevator deflection is required, but the counteracting forces will be considerable.

5. Acceleration and unsteady climb are accomplished in such a way that by the moment of achievement of a speed of 140 km/h the aircraft has climbed 20-25 m with the force on the steering wheel being reduced by means of the elevator tab. Further climb is made at a speed of 140 km/h. After obstacles are cleared (if there are none, then at an altitude of not less than 25 m) the throttle is reduced smoothly to the highest cruise rating.

#### 6. Peculiarities of Takeoff with Flaps

Takeoff with flaps is recommended for speed of headwind not higher than 10 m/s. Depending upon conditions of start, upon decision of the crew commander, takeoff can be performed with flaps at an angle of up to  $30^{\circ}$ .

Takeoff with flaps at  $30^{\circ}$  is made only with use of takeoff engine power.

Separate use of flaps is prohibited, since lift of the wings then creates torque, which lowers the strength of the wing cell.

During takeoff with deflected flaps on certain aircraft, in the middle of the run (at a speed of 50 km/h) the slats are opened and left open to moment of lift-off.

Operation of the slats on takeoff does not cause essential

change in the behavior of the aircraft, but it should be borne in mind that with emergence of the slats lift is increased, and aircraft will have a tendency to leave the ground at lower speed.

Separation of an aircraft from the ground with flaps deflected is more energetic than during takeoff without flaps. Therefore after lift-off it is necessary through proportional forward deflection of the wheel to delay departure of the aircraft from the ground.

It is necessary to consider a peculiarity in controllability of the An-2 aircraft, which consists in that during takeoff with flaps, as compared to takeoff without flaps, the amount of downward balancing deflection of the elevator is increased, and counter-acting forces decrease. Because of this, the pilot is given the false impression that during takeoff with flaps the aircraft's tendency to pitch is less. The unsteady ascent phase is accomplished in such a way that upon achievement of an altitude of 15-20 m speed is 120 km/h.

Then at a constant, ideal speed of 120 km/h climb is made to 50 m, and the flaps are retracted in 2-3 stages. Retracting of flaps in one stage and at an altitude of less than 50 m is not permissible, since the sharp decrease in lift cause the aircraft to drop, creating the possibility of collision with obstacles.

During retraction of flaps it is necessary to hold the column forward in such a way that by the moment of retraction of the flaps speed is 135-140 km/h, at which climb to assigned altitude should be made. If takeoff was made on takeoff power, after obstacles around the airfield have been cleared (if there are none, at an altitude of not less than 25 m), it is necessary to decrease power to highest cruise level, or to nominal, depending upon conditions of climb.

If the aircraft banks sharply when the flaps are retracted, which can be caused by failure of the system of linkage to one

of the flaps, it is necessary to quickly deflect the ailerons and rudder in the direction opposite the bank, hold the aircraft steady, and then immediately lower the flaps and land on the airfield. During the landing approach with flaps down speed must be held below 150 km/h and turns must be made with angle of bank not exceeding  $10^{\circ}$ .

#### 7. Peculiarities of Takeoff Under Conditions of Low Air Density

The complexity of execution of a takeoff under conditions of low air density is due to impairment of the takeoff characteristics of the aircraft, since available power decreases and required power increases, as a result of which run and takeoff distances are increased.

Thus, for example, with ambient air temperature of  $40^{\circ}\text{C}$  and pressure 700 mm Hg the altitude, according to the International Standard Atmosphere table, will be 1495 m, i.e., takeoff conditions will be the same as during flight of the aircraft at design altitude. During takeoff under these conditions at full throttle the engine will develop power close to rated power at ground level, which is less than takeoff power under conditions of the International Standard Atmosphere at 180 hp.

On the basis of this, during takeoff under conditions of low air density it is necessary:

- a) to determine run distance on the nomograph;
- b) initial position for takeoff is such that the entire length of the runway is used for the run;
- c) takeoff is made with maximum engine power (full throttle) with flaps at  $30^{\circ}$ ;
- d) in process of takeoff lift-off at low speed is not allowed; to prevent lift-off at low speed during takeoff from an uneven field,

the tail wheel should be slightly off the ground during the run, which increases the run distance somewhat.

#### 8. Peculiarities of Takeoff from Unrolled Snow Airfields and Airfields with Soft or Sandy Ground

Taxiing over an unrolled snow airfield on a wheel landing gear or an airfield with soft or sandy ground involves the danger of nosing over. In Fig. 97 it is shown that during taxiing an aircraft is acted upon by force:  $G$  — weight,  $N_p$  — ground reaction,  $F_H$  — inertia, and  $F_{Tp}$  — friction.

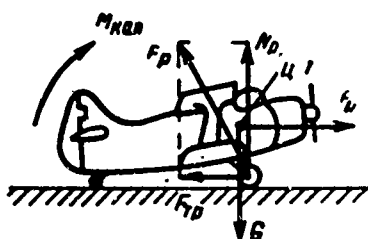


Fig. 97. Forces affecting an aircraft during sharp braking.

If taxiing is done on an airfield with smooth surface and the pilot uses the brakes competently, the slope of force  $F_p$  — the resultant forces  $F_{Tp}$  and  $N_p$  — is such that it passes ahead of the center of gravity of the aircraft, and its moment with respect to the center of gravity forces the tail of the aircraft earthward, as a result of which with the steering column back nosing over of the aircraft is prevented.

If the wheels are deeply sunken or with sharp braking the frictional force is increased so much that the resultant of the forces of reaction and friction passes behind the center of gravity of aircraft and creates a moment causing nosing over of aircraft ( $M_{Kан}$ ), as shown in Fig. 97. Furthermore, due to varying density of the snow cover in the process of the run over lightly rolled snow, the aircraft yaws, making it difficult to maintain the direction of the run and requiring raised attention of the pilot.

For safety purposes takeoffs on wheel landing gears on unrolled snow are permitted:

- a) with freshly fallen or loose snow with depth of not over 35 cm;
- b) with packed or lightly rolled snow with depth of not over 25 cm.

For takeoff from snow or soft or sandy ground it is necessary:

- a) before takeoff to adjust the center of gravity to as far aft as possible;
- b) to select the initial position for takeoff in such a way as to use the entire length of the runway for the run;
- c) to takeoff on takeoff power with flaps at  $30^{\circ}$ ;
- d) with increase of speed in the run, by smoothly pulling back the wheel, to facilitate passage of the wheels of the main landing gear over the surface of the airfield, at the same time preventing lift-off at low speed.

#### 9. Peculiarities of Takeoff with Crosswind

The presence of crosswind worsens the takeoff characteristics of the aircraft and complicates the takeoff, since asymmetric flow around the half-cells of the wings creates a difference in amounts of lift, which causes banking during the run and drift with the wind after lift-off, while asymmetry of flow around the vertical empennage and fuselage causes a turning moment windward.

Takeoff with cross wind is permitted:

- a) when the angle between wind direction and start is  $90^{\circ}$  — for wind speed of not over 6 m/s;

b) when the angle between wind direction and start is  $45^\circ$  — for speed of wind of not over 8 m/s.

During takeoff with crosswind it is necessary:

1) for complete safety and to ensure the clearing of obstacles, after receiving clearance from the control officer, to take an initial position for takeoff such that the lateral component of wind is reduced;

2) for reduction of the rolling moment during the run, to turn the steering wheel into the wind, and to turn the rudder downwind to prevent the aircraft from turning;

3) to make the takeoff on takeoff power with undeflected flaps; necessity of using takeoff power is due to the need to increase the margin of thrust in order to overcome the additional drag induced by the great deflection of control surfaces, and to increase the effectiveness at the beginning of the run as a result of airflow; the flaps are not deflected because with asymmetric flow around the half-cells of the wings, when deflected they would considerably increase the difference in values of lift on these cells and complicate takeoff;

4) with increase of speed during the run and increase of effectiveness of the ailerons and rudder, to gradually decrease their angle of deflection in order to maintain assigned direction of takeoff and accomplish lift-off without bank;

5) to accomplish lift-off upon achievement of a speed of 85-95 km/h without allowing repeated contact of the wheels with the ground, with which a lateral blow due to drift can lead to an accident;

6) after lift-off, to combat drift to an altitude of 50 m by banking windward and deflecting the rudder downwind; above 50 m drift is combated by varying the lead angle.



During takeoff with crosswind it is necessary to always be prepared for the fact that opening the slat in the wing turned windward will cause the aircraft to bank in the direction of the wing with the closed slat.

The rolling moment created with opening of the slat should be countered by energetic deflection of the ailerons, and when necessary by rudder deflection also.

## CHAPTER XII

### LANDING OF THE AIRCRAFT

#### 1. Stages of Landing and Forces Acting on the Aircraft When Landing

Landing is the retarded motion of an aircraft from an altitude of 25 m up to a full stop after a landing run.

Landing consists of five stages: descent, levelling off, holding off, landing and landing run.

Descent of the aircraft down to an altitude of the beginning of levelling off is one of the important stages in providing a normal outcome of a flight.

Statistics show that the greatest number of flight accidents occurs due to errors in piloting technique in the stage of the prelanding descent and the approach to it. The basic cause of these errors is the fact that the pilot, in distracting his attention from piloting of the aircraft to observe the land, refinement of the landing pattern and correctness of the approach along the axis of the runway, loses speed and disturbs the coordination of deflection of the controls.

Noting that at the assigned angle of bank and set speed the approach along the axis of runway will be inaccurate (usually, when the fourth turn is started late), the pilot, wishing to accelerate the turn, increases the angle of bank and, in order to accelerate

rotation, pulls control column back. The increase in angle of bank and deceleration is accompanied by a decrease in the vertical component of lift  $Y_1$  and the formation of  $\Delta G = G - Y_1$  (see Fig. 91). Under the action of  $\Delta G$  the aircraft lowers its nose and acquires motion on the lowered half-cell of the wings. Desiring to cease such motion of an aircraft, the inexperienced pilot additionally pulls back on the control lever and stalls aircraft. Here there is special danger in the disturbance of the coordination of deflection of the controls with the presence of external sideslip, at which stalling of the aircraft can occur considerably earlier. Therefore, the fourth turn must be made with special thoroughness, strictly maintaining the speed and angle of bank, controlling the coordination of deflection of the controls by the sideslip indicator.

After fulfillment of the fourth turn up to the beginning of levelling off, descent should be produced at constant speed and the relationship of forces acting on the aircraft the same as those during steady descent (see Fig. 89).

With refinement of the landing pattern and deflection of the flaps one should consider the following:

1. With deflection of the flaps the An-2 aircraft pitches up. At the time of deflection of the flaps the aircraft is "tossed" upwards due to the rapid increase in lift with almost constant speed of flight during the time of the lowering of the flaps.

As a result of the formed pitching the angle of attack will be increased, and therefore the reserve lift appearing with deflection of the flaps will increase even more. The formed reserve lift is a centripetal force which distorts the trajectory upwards, and the aircraft can lose speed. The loss in speed in this case will occur because of a decrease in the component of weight  $G_2$  and increase in coefficient  $c_x$  due to the deflection of the flaps and increase in angles of attack under the action of an unbalanced positive pitching moment.

To prevent the loss in speed it is necessary at the time of deflection of the flaps to push the control wheel forward and then, setting the angle of descent, corresponding to the descent of an aircraft with deflected flaps at a speed of 120 km/h, remove the force from the control column by the elevator trim tab.

It is impermissible to produce a refinement of the landing pattern during flight by retracting the lowered flaps (as this is sometimes practiced). Retracting of the flaps is accompanied by a sharp increase in vertical descent rate and increase in angle of attack, as a result of which the aircraft approaches stalling.

2. Correction of the landing pattern with an undershoot owing to the augmentation of thrust during landing with undeflected flaps usually leads to a certain trajectory acceleration; if, however, the flaps are deflected, then acceleration, as a rule, does not occur. Therefore, control of the engine during descent with undeflected and deflected flaps should be different. This distinction consists in the following:

1) in the case of short-term approach with deflected flaps it is necessary to throttle the engine at high altitude in order to have time to restore the initial angle of descent, since otherwise the aircraft will lose speed. If the throttle valve is covered late, then in the view of the proximity of the land there will not be sufficient reserve of speed for levelling off, since after retracting of the supercharging the given angle of descent will correspond to large angles of attack. After the retracting of supercharging the aircraft acquires additional vertical velocity of descent, which increases the angle of attack with maintaining of the previous direction of longitudinal axis of the aircraft; this can lead to a release of the slats with the beginning of levelling off, and the aircraft from a high altitude will start to pancake and, reaching considerable vertical velocity will hit against the land with the wheels;

2) during the approach with undeflected flaps the trajectory changes insignificantly, and the angle of attack after retracting of the supercharging will be changed also insignificantly; therefore, supercharging can be removed later as compared to the case of the approach of the aircraft with deflected flaps;

3) in the case of a steady descent for landing with deflected flaps on increased supercharging, the engine should be throttled later, since premature retracting of supercharging in this case at high altitude from land will lead to rapid deceleration of the aircraft in speed and pancaking with lowering of the nose.

In all cases of sharp deceleration on the section of prelanding descent, one should smoothly increase the supercharging and depart for a second circle.

Levelling off occurs along a curvilinear trajectory, which constitutes the transition from a rectilinear slanted trajectory of descent to a trajectory of horizontal flight at the end of the levelling off. In the approach to the altitude of levelling off, the pilot, pulling back on the control column, increases the angles of attack, as a result of which there is created a surplus of lift  $\Delta Y$ , which is equal to the total lift  $\Delta Y = Y - G_1$ . With an increase in the angle of attack there is increased the resisting force  $Q$ , and from the decrease in slope of the trajectory the component of weight  $G_2$  decreases, which is for overcoming the drag from which the speed for levelling off decreases (Fig. 98a).

Holding off is produced for deceleration prior to landing. Figure 98b shows that during holding off the aircraft flies horizontally ( $Y = G$ ), and motion of the aircraft delayed, since  $Q$  is unbalanced.

To ensure equalities  $Y$  and  $G$ , it is necessary in conformity with a decrease in the square of the speed to increase the coefficient  $c_y$ , for which the pilot with deceleration increases the angle of attack, pulling back on the control column.

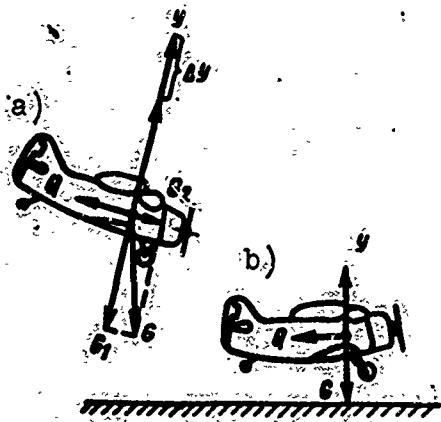


Fig. 98. Forces acting on the aircraft with leveling off and holding off.

During landing of the aircraft with deflected flaps, due to the shielding action of the land, downwash along the horizontal empennage during holding off decreases. In the second half of the holding the positive pitching moment of the horizontal empennage completely vanishes, and the aircraft has a tendency to decrease the angle of attack, which requires a more intense pulling back of the control column.

Landing is the cancellation of speed prior to a landing pancake of the aircraft at the end of holding off before the moment of contact of wheels of the aircraft on the runway. Figure 99a shows that due to the increase in vertical descent velocity ( $V_y$ ) with a pancake landing the true angle of attack ( $\alpha_{\text{act}}$ ) becomes larger than the geometric ( $\alpha$ ). With an insignificant error of the pilot in the determination of altitude of the end of levelling off, which is governed by the altitude of landing of the aircraft, this presents no special danger for the An-2 aircraft.

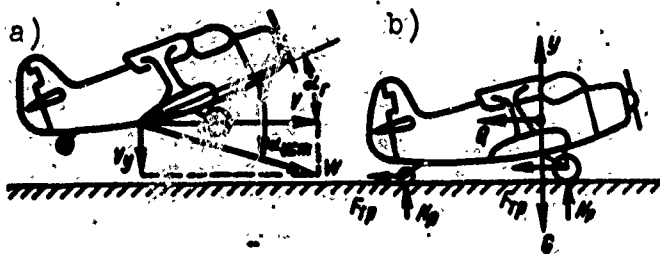


Fig. 99. Landing of the aircraft and forces acting on the aircraft with a landing run.

The An-2 aircraft has a landing angle of attack of the upper wing of  $15^\circ$ ; during landing with deflected flaps because of the release of the slats at the end of levelling off the critical angle of attack of it will be increased, which eliminates stalling of the aircraft during landing.

However, one should consider that in the case of a gross error in the levelling off the aircraft, which is in a three-point position, will pancake from a high altitude, and the true angle of attack of it can exceed the critical. In spite of the good lateral stability and controllability of the aircraft at large angles of attack with the appearance of banks at the time of landing of the aircraft, especially during landing by pancaking it is necessary to correct them by powerful deflection by the rudder and ailerons.

The landing run is a rectilinear section of the motion of the aircraft along the runway from the moment of contact of the wheels with land up to a full stop. On the landing run, besides aerodynamic forces  $Y$  and  $Q$ , acting on the aircraft are the force of reaction of the land  $N_p$  and frictional force of the wheels against the surface of the airfield  $F_{\mu}$  (Fig. 99b). With the beginning of deceleration the frictional forces sharply increase. With deceleration lift ( $Y$ ) and drag ( $Q$ ) decrease, and the forces of reaction of the land are increased. To avoid turns on the landing run it is necessary thoroughly to maintain the rectilinearity of the run, for which the pilot must select reference point on the horizon and look at it after landing of the aircraft.

## 2. Landing Characteristics of the Aircraft

Landing is characterized by landing speed, landing run and length of the landing distance. The speed of the aircraft at the time of contact of its wheels on the runway is called landing speed

$$V_{\text{loc}} = 0,94 \sqrt{\frac{2G}{c_{y_{\text{loc}}} S}}.$$

where 0.94 — coefficient considering the deceleration in the process of pancaking due to the braking action of the land.

The path passable by the aircraft from the instant of touchdown to its full stop is called the landing run

$$L_{app} = \frac{(V_{acc} \pm a)^2}{2j}$$

The path passable by the aircraft with respect to land in the descent of the aircraft from an altitude of 25 m to a full stop on the landing run is called the landing distance

$$L_{acc} = (2.5 \div 3) L_{app}$$

During landing with a landing weight of 5250 kg on an airfield with a hard grassy cover during a calm with the application of brakes, the An-2 aircraft has the following landing characteristics (Table 6):

Table 6.

Form of landing	Landing speed, km/h	Landing run, m	Length of landing distance, m
With undeflected flaps..	110	350-400	750-800
With deflected flaps....	85-90	200-250	550-600
From pancaking.....	65-70	95-100	270-300

In the case of failure of the brake system the landing run is increased 100-150 m.

The length of the landing run of the aircraft is determined by nomograph No. 2 by the same method as takeoff distance.

### 3. Landing with Deflected Flaps

The basic form of landing in the An-2 aircraft is landing with flaps, deflected at an angle of  $30^\circ$ , the opening of slats at the end



of levelling off and landing of the aircraft on three points at a steady speed without drift. Landing with deflected flaps is accomplished with a headwind of not more than 18 m/s.

With the fulfillment of training flights by students it is recommended to deflect the flaps with a wind speed up to 8 m/s -  $30^{\circ}$ , with a wind speed of 8 to 10 m/s -  $15^{\circ}$ , and with a wind of more than 10 m/s - without deflection of the flaps.

The landing pattern and landing is made by the commander of the crew; the copilot controls the operation of the engine and regulates the temperature rate and also observes the right hemisphere.

In landing the commander of the crew is obliged to do the following:

1. Along a tangent to the nearest turn at an altitude of 300 m with a speed of 150 km/h (supercharging of 600 mm Hg, revolutions 1600 r/min) enter into a circle.

2. With the approach to the third turn decrease the speed down to 140 km/h and with an angle of bank up to  $30^{\circ}$  make the third turn: with a headwind of up to 5 m/s the beginning of the turn is made after flying by the landing "T" by a width of the wing and with a wind of 10 m/s - after flying by the crossbeam "T." The magnitude of the turn is determined by the force of the wind and is controlled by ground reference points and the directional compass.

3. After the third turn, when the line of landing signs are projected on 1/3 of the average glass of the canopy of the pilots, turn to conditions of descent: speed along the trajectory, 140 km/h, vertical descent velocity 1.5 m/s, (manifold pressure 370-400 mm Hg).

After decreasing the supercharging decrease the pulling forces on the control wheel by deflecting the elevator trim tab.

4. When the angle between lines of the landing signs and lines of sighting on the landing "T" is  $12-13^{\circ}$ , at an altitude of 180-200 m put the aircraft in a fourth turn, fulfilling it at a constant by speed of 140 km/h and with a bank up to  $30^{\circ}$  and paying special attention to maintaining the speed and coordination of deflection of the controls.

Starting the turn with a bank of  $30^{\circ}$ , gradually decrease the angle of bank in the course of a turn in order to ensure recovery from the turn in parallel to the axis of the runway, and at an altitude of not lower than 100 m. In the case of deflection of the aircraft from the axis of the runway at an angle up to  $10^{\circ}$ , correct the approach by an additional turn of the aircraft, and with great deviation depart for the second circle.

5. After coming out of the fourth turn change the propeller to "low pitch," decrease the manifold pressure to 300 mm Hg, set the speed at 130 km/h and refine the calculation with respect to the point of levelling off, which must be outlined at a distance up to the "T":

With wind up to 5 m/s.....	after 180 m
With wind up to 7 m/s.....	after 150 m
With wind up to 10 m/s.....	after 100 m

At an altitude of not less than 100 m decrease the speed down to 130 km/h and deflect the flaps. Correct the tendency of the aircraft to pitching with deflection of the flaps by deflection of the control column. After deflection of the flaps set the rate of descent at 120-125 km/h and remove the load from the control wheel by the trim tab.

6. Carry out descent down to an altitude of the beginning of levelling off at a constant speed equal to 120-125 km/h, from an altitude of 30 m look at the land along the left side for the purpose of determining the altitude of the beginning of levelling off. At an altitude of 10 m smoothly throttle the engine and transfer the right hand to the control wheel.

7. From an altitude of 6-7 m smoothly pull the control column back and start to level the aircraft with such calculation in order to complete levelling off at an altitude of 0.7-0.8 m and to ensure opening of the slats at this height. With levelling off of the aircraft look forward at a distance from the leading edge of the wing of 25-30 m and to the left of the axis of the aircraft by 20-25°.

With release of the slats at the end of levelling off, the aircraft starts to pancake, in the process of which in proportion to the approach of the aircraft to land it is necessary to pull the control column back by such calculation in order to put the aircraft in a three-point position at an altitude of 0.25-0.3 m.

8. After landing the aircraft look at the horizon and, holding the control column back maintain the direction of the landing run on the outlined reference point: in the first half — by the rudder and in the second — by smooth application of the brakes.

#### 4. Peculiarities of Landing Without Deflection of the Flaps

Landing without deflection of the flaps is produced with a defective control system of the flaps (students in schools land with a wind of more than 10 m/s).

With landing of the aircraft without the application of flaps one should do the following:

1) descend after the fourth turn with a speed of 130-135 km/h (manifold pressure 350 mm Hg); here the trajectory of descent is considerably more sloping than in landing with deflected flaps, which worsens the view from the aircraft forward and to the right and requires increased attention of the crew;

2) throttle the engine with the beginning of levelling off, and level off at an altitude of 4-5 m and finish at an altitude of 0.5-0.7 m;

3) holding off with such a calculation in order give to the aircraft a landing angle of attack at an altitude of 0.25-0.3 m and pull control column back more smoothly than during landing with deflected flaps.

#### 5. Peculiarities of Landing on Main Wheels

Landing on the main wheels is one of the forms of normal landing of the aircraft and the following is recommended:

- a) with a headwind — more than 15 m/s;
- b) when the pilot is not sure in making a normal three-point landing because of the low approach of the aircraft to the land during landing without flaps;
- c) when there is no confidence in the exact determination of distance to the land (with snow cover, under conditions of rain and snowstorm, and at twilight).

During landing of the aircraft on the main wheels and with strong headwind the following is expedient:

- 1) produce descent of the aircraft down to an altitude of the beginning of levelling off without deflected flaps at a speed of 160 km/h;
- 2) start levelling off of the aircraft at an altitude of 4-5 m and finish at an altitude of 0.3 m;
- 3) do not allow soaring up of the aircraft on holding off. At the time of contact of wheels of the aircraft on land by an insignificant deflection of the control column forward prevent separation of the aircraft from the land.

During landing of the aircraft on the main wheels in those cases when it is difficult to determine exactly the distance to the land, one should:

1) produce descent of the aircraft with deflected flaps at a speed of 120-125 km/h;

2) start levelling off of the aircraft from an altitude of 6-7 m by a smooth delayed deflection of the control column back up to the moment when the surface of the land will appear clearly.

In all cases with deceleration on the landing run pull the control column back and only when it will be completely pulled back, apply the brake.

#### 6. Peculiarities of Landing with a Cross Wind

In descending with a cross wind the aircraft crabs and drifts downwind, and after landing it turns upwind. Landing with a cross wind in the An-2 aircraft is permitted under the following conditions:

a) at an angle between the direction of the wind and start of  $90^\circ$  - not more than 6 m/s;

b) at an angle between the direction of the wind and start of  $45^\circ$  - not more than 8 m/s.

Landing with a cross wind is produced with undeflected flaps for the same reason as that of takeoff.

In the case of an emergency (landing on limited site) it is permitted to use flaps during a cross wind with a speed of not more than 3-4 m/s.

With the fulfillment of landing with a cross wind the following is necessary:

1. Consider the effect of the cross wind in carrying out the landing approach: with the left circle and wind on the left start the fourth turn earlier, and with the wind on the right - later.

2. Accomplish descent of the aircraft after the fourth turn down to an altitude of the beginning of levelling off at a speed of 140 to 145 km/h.

3. Combat drift by a combined method: up to an altitude of 50 m — by a lead angle, from an altitude of 50 m — by the creation of bank upwind and by holding off of the direction of the flight — deflection of the rudder downwind. Maintain the constancy of the angle of bank (by the magnitude of drift) up to the altitude of the beginning of levelling off.

4. Start levelling off of the aircraft from an altitude of 4-5 m and finish at an altitude of 0.5 m. With the beginning of levelling off of the aircraft gradually decrease the bank with such calculation in order to ensure landing of the aircraft without a bank and on three points. Landing of the aircraft on three points will considerably facilitate the holding of the direction of the landing run. With drift at the time of the landing of the aircraft, to decrease the lateral impact load on the landing gear and to prevent a turn upwind, deflect the rudder toward the drift, conforming with the force of the drift (Fig. 100).

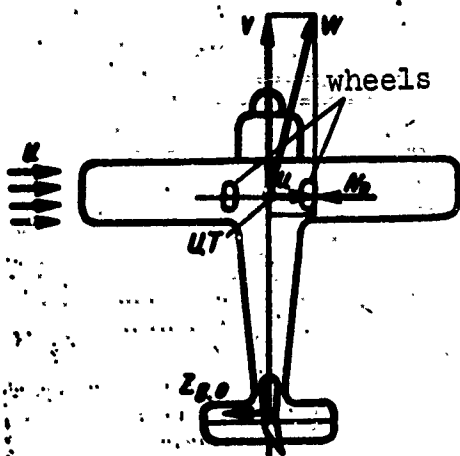


Fig. 100. Forces acting on the aircraft with rudder deflection at the time of landing when landing with a cross wind.

5. After landing the aircraft maintain the direction of the landing run: in the first half - by the rudder, deflecting it into the wind, and by the ailerons, turning the control wheel upwind; in the second half of run - application of the brakes.

## 7. Peculiarities of Landing on an Unrolled Snow Airfield

In landing on an unrolled snow airfield there is the danger of a nose-over of the aircraft. Landing is permitted with a depth of cover of snow up to 35 cm and is produced as a landing of normal profile. At the moment of landing of the aircraft the control column should be energetically and completely pulled back. Brakes on the landing run are not used, with the exception of cases of an emergency.

In the landing approach create centering as much as possible, not exceeding the bounds of that established.

## 8. Peculiarities of Pancake Landing

Pancake landing is applied in exceptional cases when landing on an unfamiliar site with obstacles in the approach zone to it. Pancake landing requires especially increased attention and definite actions.

Transition to pancaking is produced after overcoming obstacles in conditions of horizontal flight from an altitude of not less than 100 m.

Pancake landing can be produced only with the operation of the slats or with the joint work of the slats and flaps.

In carrying out a pancake landing the following is necessary:

1) descend after transition to pancaking: with operation of the slats — at a speed of 105-110 km/h, and with joint operation of the slats and flaps — at a speed of 90-95 km/h; remove pulling forces on the control column appearing with a release of the slats by the elevator trim tab;

2) do not allow an excess in the rate of descent along the trajectory on which slats operated more than 5 km/h, and an increase in supercharging with refinement of the calculation — more than

500 mm Hg; if the parameters shown are not maintained the slats are extended toward the leading edge of the wing and conditions of pancaking are disturbed;

3) start levelling off of the aircraft from an altitude of 8 m; with the beginning of levelling off increase the supercharging up to 500 mm Hg, which is necessary for decreasing of vertical rate of descent and increasing the effectiveness of elevator owing to its air flow created by the propeller;

4) pull back the control column in the process of levelling off by such a rate in order to complete levelling off of the aircraft at an altitude of 0.5 m; decrease the supercharging with such calculation so that prior to the moment of creation to the aircraft of a three-point position it will be completely removed.

#### 9. Errors in Landing, Their Causes and Methods of Correction

The most widespread errors made by pilots in landing are high levelling off, soaring up, and landing of the aircraft with subsequent separation from the ground. The given errors appear because of nonfulfillment by the pilot of parameters of landing with respect to speed and altitude set by the Manual on Flying Operation of the An-2 Aircraft or incorrect distribution of attention.

1. High levelling off occurs because of the incorrect determination of the altitude of the beginning of levelling off, sharp deflection of the control column, incorrect direction of the view in the process of levelling off -- nearer than 20 m from the leading edge of the wing or at an angle from the longitudinal axis of the aircraft of more than  $25^{\circ}$ .

For the correction of high levelling off one should retard the control column and bring the aircraft down to an altitude of 0.7 to 0.8 m and then in proportion to the approach of the aircraft to the land pull control column back, producing a three-point landing.



2. Soaring up is a result of distraction of the look of the pilot from the land, putting the aircraft at an altitude of the beginning of levelling off at a speed considerably exceeding that for the given form of the landing, and also the disproportionate deflection of control column back with descent of the aircraft on holding.

Soaring is corrected by the smooth proportional deflection of the control column forward for cessation of the departure of the aircraft from land and then similar to the correction of high levelling off.

3. The causes of landing of the aircraft with subsequent separation from the land are:

a) the ignorant fulfillment by the pilot of landing on the main wheels, when instead of holding the control column at the time of landing the pilot deflects it forward.

b) energetic deflection of the control column back at the time of landing of the aircraft on the main wheels when landing without flaps, especially in a strong wind;

c) disproportionate deflection of the control column to the approach of the aircraft to land, which is characteristic for young pilots in the landing of an aircraft with a small reserve of fuel without loading (after fulfillment of aerial spraying and dusting) when the center of gravity limits of the aircraft becomes more forward;

d) low levelling off of the aircraft because of the incorrect direction of the view -- further than 30 m from the leading edge of the wing or at an angle from the longitudinal axis of aircraft of less than  $20^{\circ}$ .

With correction of an error it is necessary:

1) to retard the steering wheel, not deflecting it forward, since the An-2 aircraft departs from land after contact by the wheels at an insignificant altitude, and deflection of the control column forward with separation of the aircraft from land almost without speed can lead to the fact that the aircraft repeatedly will hit its wheels against land at such an angle at which nose-over is possible;

2) with the approach of the aircraft to the land, to produce a three-point landing by the proportional deflection of the control column back.

The opening of slats on the An-2 aircraft ensures the relatively soft landing of the aircraft and great reserve of lateral stability, in view of which there is no need with any soaring up of the aircraft to depart for a second circle.

Deflect the control column back by a more energetic motion to give to the aircraft a landing position with descent from an altitude of the beginning of holding off. Eliminate the banks formed by an energetic deflection rudder and ailerons with the immediate setting of them in a neutral position with the beginning of motion of the aircraft in the initial position.

#### 10. Departure for a Second Circle

The available power of the power system of the An-2 aircraft ensures the possibility of departure for a second circle from descent of the aircraft with undeflected and deflected flaps and also from pancaking with simultaneous operation of the flaps and slats.

Departure for a second circle from descent with undeflected flaps is possible at any stage of descent up to the moment of levelling off of the aircraft with use of the rated power of the engine.

Departures for a second circle with deflected flaps from pancaking is permitted from an altitude of not less than 10 m; it requires the use of takeoff power of the engine, definite actions and increased attention of the pilot.

In using takeoff power of the engine with deflected flaps at an angle of  $30^{\circ}$  and, even more, at an angle of  $40^{\circ}$  because of lower decentering and great downwash angle on the horizontal empennage, on the aircraft there appears considerable positive pitching moment, which with ignorant actions of the pilot can lead to the loss in speed and stalling of the aircraft.

With departure for a second circle the following is necessary:

1) with low altitude, not looking away from land, increase the power of the engine and with gradual departure from land produce climb at these speeds:

With undeflected flaps..... 140 km/h

With deflected flaps..... 120-125 km/h

2) if departure for a second circle is produced with deflected flaps, maintaining the indicated constant speed, climb to 50 m and in a two-three procedure retract the flaps:

3) after retracting the flaps at an altitude of 50 m (with undeflected flaps — at an altitude of 25 m) set the cruise setting (supercharging, 760 mm Hg, revolutions, 1850 per minute), then at a speed of 140 km/h gain altitude and carry out a repeated landing approach.

To ensure safety of the departure for a second circle one should:

1) not allow a sharp increase in power of the engine, since this can cause failure of engine and intense increase in the positive pitching moment;

2) parry the appearing positive pitching moment with an increase in power of the engine by deflection of the control column forward, removing the appearing forces on the control column by the trim tab. In the case of sharp pitching of the aircraft, which it is impossible to cease even with full deflection of the control column forward, carry out recommendations discussed in Chapter V, Section 11;

3) control the speed of flight according to the position of the aircraft with respect to the horizon, since the speed indicator lags and readings of it do not correspond to angles of attack.

At a speed of 140 km/h the horizon is projected under the fairing of the air inlet and with ascent with deflected flaps at a speed of 125 km/h — along the upper section of the fairing of the air inlet.

## C H A P T E R XIII

### PECULIARITIES OF FLIGHT OF THE AIRCRAFT IN CONDITIONS OF ICING

#### 1. Character, Degree and Intensity of Icing

Icing of the aircraft is one of the meteorological phenomena dangerous for flights, since it considerably lowers the flight safety.

Icing of the aircraft is accompanied by impairment of its aerodynamic and flying characteristics dependent on the character, degree and intensity of icing. The cause of icing is the presence in the atmosphere of vapor, supercooled drops of water and crystals. Icing can be in the form of hoarfrost, ice and rime.

Hoarfrost is a thin crystal deposit forming on the surface of the aircraft when it moves from colder to less cold layers of the atmosphere.

Ice will be formed in the presence in the atmosphere of supercooled drops of water. The basic varieties of ice are: transparent ice (glaze) and frosted ice.

Transparent ice (glaze) will be formed during flight in clouds containing big drops at temperatures from 0 to  $-5^{\circ}\text{C}$  and is deposited on the leading edge of the wing in the form of an uneven trough-shaped growth, spreading along the wing at a width up to 300 mm and more.

During flight in a zone of freezing rain ice can cover the whole surface of an aircraft.

Opaque ice (frosted) will be formed during flight in clouds containing small drops and ice crystals, in the range of temperatures from 0 to  $-10^{\circ}\text{C}$ , and is deposited on the narrow section of the leading edge of the wing in the form of a wedge-shaped growth.

Rime is variety of frosted ice, and has a brightly marked needle-shaped crystal character.

Icing is characterized by degree and intensity. Degree of icing means the quantity of ice in millimeters deposited on the surface of the aircraft.

Intensity of icing is called the quantity of ice deposited on the leading edge of the wing in the middle of semispan of it in one minute. The intensity of icing is subdivided into weak — 0.5 mm/min, medium — up to 1 mm/min, heavy — over 1 mm/min.

The degree of icing depends on the duration of stay of the aircraft in the zone of icing and thickness of the profile: the longer the aircraft is in the zone of icing, and the less the thickness of the profile, the greater the degree of icing.

The intensity of icing depends on the speed of flight of the aircraft and thickness of the profile. The thicker the profile, the greater the local velocity of the flow around it, as a result of which the drops spread greater over the surface, and the intensity of icing decreases. The increase in flight speed of a low-speed aircraft is accompanied by an increase in intensity of icing, since the aircraft encounters greater quantity drops per unit time on its path.

The degree and intensity of icing can be determined in flight with the help of mechanical indicators and by the deceleration of flight. Deceleration of the aircraft by 10-15 km/h during 5 minutes of flight in the zone of icing in constant operating conditions of the engine indicates the great intensity of icing.

The most dangerous form of icing, characterized by the greatest intensity and considerable distortion of the form of profile of the wing, is glaze.

## 2. Influence of Icing on the Aerodynamic Characteristics of the Aircraft

With any form of icing the aerodynamic properties of the aircraft are considerably worsened, since icing distorts the form of the profile and increases the roughness of the surface of the aircraft.

With icing the lift of the wing decreases because of a decrease in local velocity of flow around of the surface of the wing because of a decrease in curvature and increase in roughness.

Deceleration of flow in the boundary layer on the upper surface of the wing leads to an earlier local separation of flow with an increase in angle of attack and to a decrease in coefficient  $c_{y_{max}}$ .

With further increase in the angle of attack there occurs a general separation of flow from the wing and much earlier and more suddenly than on a clean wing. The influence of icing on the change in coefficient  $c_y$  at different angles of attack is shown in Fig. 101. In the figure it is clear that  $c_y$  of the iced wing decreases at all angles of attack, and the decrease in it is increased with the approach to the critical angle of attack, which in turn also decreases somewhat.

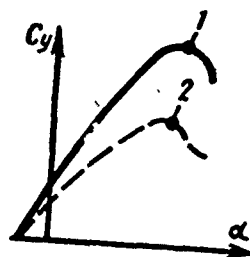


Fig. 101. Change in coefficient of lift of an iced wing: 1 — separation of flow on a clean wing; 2 — separation of flow on an iced wing.

A decrease in coefficient  $c_{y_{max}}$  leads to an increase in the speed at which general separation from the wing occurs:

$$V_{cr} = 3,6 \sqrt{\frac{2l}{c_{y_{max}} S}}$$

If the stalling speed with a clean wing for the aircraft An-2 with a flying weight of 5250 kg with extended slats is equal to 96 km/h, then depending upon the character of icing it can be considerably increased.

With icing of the aircraft in the form of frosted ice, the profile drag of the aircraft is increased basically owing to friction drag, especially at small angles of attack (Fig. 102).

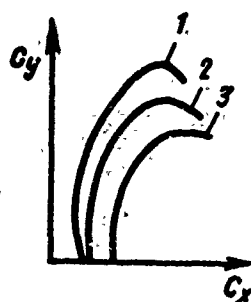


Fig. 102. Influence of the character of icing on aerodynamic properties of an aircraft. 1 - clean aircraft; 2 - icing of an aircraft in the form of frosted ice; 3 - icing of an aircraft in the form of glaze.

With icing in the form of glaze the profile drag is considerably increased in the whole angular region of attack both because of friction drag and because of pressure drag, part of which is great even at small angles of attack.

By experimental means it has been established that at angles of attack close to the optimum the coefficient  $c_x$  of an iced aircraft as compared to a clean aircraft with glaze is increased: with the degree of icing of 25 mm - by 30% and with the degree of icing of 50 mm - by 70%.

A decrease in lift and sharp increase in drag causes a sharp decrease in the lift-drag ratio of the aircraft. If the An-2 aircraft would be subjected to icing in the form of glaze with the degree at 25 mm, then the maximum efficiency of it would not exceed the efficiency of a clean aircraft with deflected flaps at an angle of  $40^\circ$ .



To create lift equal to the weight of the aircraft, a steady flight of the iced aircraft at the same speed as in normal conditions should be produced at a large angle of attack.

Since the critical angle of attack of an iced aircraft decreases, the iced aircraft considerably approaches stalling conditions, which consist of the basic danger of flight on an iced aircraft.

### 3. Influence of Icing on Flying Characteristics of the Aircraft

Icing causes sharp impairment of flying characteristics of the aircraft, since the available power decreases, and the required power is increased.

The decrease in available power is caused by icing of the blades of the propeller, as a result of which its efficiency decreases, and by icing of diffusers of the carburetor, which causes a decrease in effective power of the engine.

An increase of required power occurs because of the increase in flying weight of the aircraft by the magnitude of deposited ice and decrease in lift-drag ratio. An analysis of curves of available and required powers (Fig. 103) shows that for an iced aircraft as compared to a clean aircraft the necessary speeds at all characteristic angles of attack  $\alpha_{yk}$  and  $\alpha_{st}$  are increased, the maximum speed ( $V_{max}$ ) and surplus of power ( $\Delta N$ ) decreases, as a result of which the speed range is reduced and the vertical ascent velocity decreases. A decrease in vertical ascent velocity leads to a decrease in the ceiling of the aircraft, and an increase in necessary thrust, and required power and a decrease in efficiency of the propeller lead to a decrease in distance and duration of the flight.

With icing of the aircraft in the form of glaze with the degree of 25 mm, flying characteristics of it considerably worsen, which will be thus expressed:

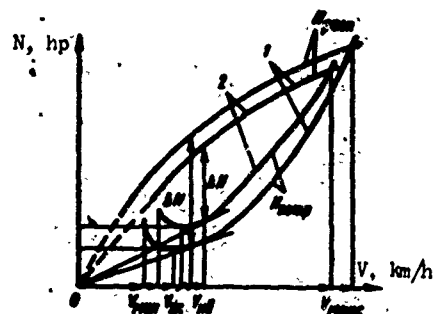


Fig. 103. Influence of icing on flying data of the aircraft. 1 — clean aircraft; 2 — iced aircraft.

- a) in the increase in minimum necessary speed and decrease in maximum speed by 20-25%;
- b) in the decrease in vertical ascent velocity by 35-40%;
- c) in the decrease in ceiling of the aircraft by 30%;
- d) in the decrease in distance and duration of flight by 15-20%.

The impairment of aerodynamic and flying characteristics of the aircraft with icing sharply lowers its piloting properties, in view of which flight under conditions of icing on the An-2 aircraft, not having de-icers on the wing and empennage, is prohibited.

There is considerable danger also in takeoff on an iced aircraft.

With takeoff on an iced aircraft the speed on the takeoff run is increased more slowly because of the increasing drag induced by the deposit of ice and the increased friction of wheels against the land due to a decrease in lift of the iced wing and therefore the takeoff distance is sharply increased.

Therefore, upon achievement by the aircraft of a set lift-off speed, with an increased takeoff distance, the pilot during takeoff can worsen the error by trying to pull the control column back to accelerate lift-off.

An increase in the angle of attack during lift-off can appear so close to the critical that even if the aircraft leaves the ground,

then an insignificant motion of the control column or the effect of the flow of air are enough to cause premature separation of flow from the wing.

The same danger is present in the overcoming of obstacles in the zone of takeoff, since a flat trajectory of ascent and increase in takeoff distance can compel pilot to increase the angle of attack.

It is necessary to consider that the character of icing of the aircraft on land is considerably more dangerous than icing in flight, inasmuch as it can be asymmetric and cause aerodynamic asymmetry. Proceeding from this, with preparation for flight it is necessary to clean the ice, snow and hoarfrost thoroughly from the surface of the aircraft.

If in flight it is not possible to avoid icing, one should make a force landing, during which:

a) do not allow sharp operation by the control vanes and control lever of the engine; cleaving of ice on the propeller blades should be produced by a change in the pitch;

b) make the turns with a bank of not more than  $10^{\circ}$  and at a speed of not less than 160 km/h;

c) descend for landing at a speed of 145-150 km/h with undeflected flaps;

d) throttle the engine only at the beginning of levelling off.

## CHAPTER XIV

### OVERLOADS AND STRENGTH OF THE AIRCRAFT

#### 1. General Information

Under different conditions of flight the construction of the aircraft undergoes a load of three forms: load from weight of the aircraft, on loading with flight in bumpy air (bumpiness) and load during fulfillment of a maneuver.

Under the action of the load elements of construction of the aircraft are deformed, i.e., they change their initial dimensions and form. Deformations can be elastic and residual.

Elastic deformations are those which vanish after cessation of the action of external forces on the aircraft. Permanent deformations are those which remain on the elements of construction of the aircraft after cessation of the action of external forces.

The ability of an aircraft to maintain external loads acting on it in flight without destruction and the appearance of permanent deformations is called strength of the aircraft.

The initial data for determining breaking loads of the aircraft are strength norms, which set the classification of the aircraft and calculation cases determining the biggest loads of main parts of the aircraft. Determination of the loads is produced taking into account the assignment of the aircraft, its flying weight and maximum flight speed.

In strength norms aircraft are subdivided into three classes:

Class "A" — maneuvering aircraft;

Class "B" — limited maneuvering aircraft;

Class "C" — nonmaneuvering aircraft, on which it is not permitted to carry out aerobatic maneuvers.

Of all the possible loads acting on the aircraft, for calculation of construction for strength the greatest loads which can take place in the operation of a given aircraft are selected. By strength norms seven basic calculation cases are provided: A, A<sup>1</sup>, B, C, D, D<sup>1</sup> and E. Added to these designations sometimes are subscripts, which indicate to what part of the aircraft pertains the given calculation case, for example A<sub>sp</sub> — case A for the wing.

Case A — curvilinear flight at an angle of attack at which the coefficient of lift will have a maximum value.

This case takes place with recovery of the aircraft from gliding, energetic transfer of the aircraft to a climb, and the effect on the aircraft of powerful vertical flows.

The An-2 aircraft as a heavy transport aircraft with a flying weight of 5250 kg and maximum horizontal speed of 250 km/h belongs to class "B." The calculation of strength of the aircraft is performed according to case A<sub>sp</sub>.

For the purpose of preventing the appearance on the aircraft of permanent deformations during its operation limitations of the indicated air speed are set:

a) lowering of flaps during landing — 130 km/h;

b) flight with deflected flaps — 150 km/h;

- c) with descent in a calm atmosphere — 220 km/h;
- d) with descent in bumpy air — 190 km/h;
- e) in horizontal flight — 250 km/h ( $q_{\text{max}} = 30$  kgf/m<sup>2</sup>).

The maximum permissible rate of descent of the aircraft with respect to conditions of strength is 320 km/h ( $g_{\text{max}} = 500$  kgf/m<sup>2</sup>).

## 2. Overloads and Coefficients Characterizing the Safety of Flight and Strength of the Aircraft

In steady flight all forces acting on the aircraft are mutually balanced.

For a change of conditions of flight it is required to change the relationship of forces acting on the aircraft, which is accompanied by the acceleration or deceleration of motion of the aircraft during which it undergoes overload.

An overload is ratio of resultant aerodynamic forces and tractive force to the flying weight of the aircraft

$$n = \frac{N}{G},$$

where  $N$  — resultant of the full aerodynamic force and tractive force.

In all cases of rectilinear steady motion of the aircraft, the resultant force ( $N$ ) is equal to the flying weight ( $G$ ) and is directed upwards vertically, and the overload ( $n$ ) is equal to one.

In those cases when the resultant force ( $N$ ) does not coincide in magnitude or direction with the gravity of the aircraft, the equilibrium of forces is disturbed and for the aircraft acceleration appears in the direction of the unbalanced part of force  $N$ .

In the general case of spatial motion of the center of gravity of the aircraft the overload is decomposed into components along axes

of the aircraft and is respectively called:

- a) longitudinal ( $n_x$ ) acting in parallel to the velocity vector:

$$n_x = \frac{P-Q}{G};$$

- b) normal ( $n_y$ ) acting perpendicular to the velocity vector in the plane of symmetry of the aircraft:

$$n_y = \frac{Y}{G};$$

- c) lateral ( $n_z$ ) acting perpendicular to the longitudinal and normal overloads:

$$n_z = \frac{Z}{G}.$$

Since the loads acting along axes X and Z are small, then overloads  $n_x$  and  $n_z$  in their magnitude represent no special danger.

From the point of view of safety one should especially discuss the normal overload ( $n_y$ ), which is the ratio of lift (Y) to weight of the aircraft (G).

In a steady horizontal flight  $n_y = 1$ .

If in a steady horizontal flight the control column is energetically pulled back, the angle of attack will be rapidly changed, and the change in speed will occur with considerable delay. The magnitude of the overload appearing with this can be determined:

$$n = \frac{Y_{n.p.n}}{Y_{r.n}},$$

where  $Y_{n.p.n}$  — lift in new conditions of flight;  $Y_{r.n}$  — lift in horizontal flight.

If with the flight of the An-2 aircraft in conditions of maximum speed ( $c_y = 0.23$ ) by sharp deflection of control column back to

transfer the aircraft into a critical angle of attack, which with extended slats corresponds to the coefficient  $c_y = 1.59$ , then there will be created the following overload:

$$n = \frac{1.59}{0.23} = 6.9.$$

To create such an overload it will be required to use great forces which the pilot, as a rule, does not have.

An overload at which it is possible to bring out the aircraft has a limit conditioned by flight safety.

#### Reserve Limit of the Overload

An overload appearing in flight is limited first of all by the critical angle of attack. The definite maximum possible overload corresponds to each of the conditions of horizontal flight.

An overload corresponding to the exit of the aircraft into the critical angle of attack from conditions of horizontal flight is called available overload:

$$n_{\text{расч}} = \frac{c_{y_{\text{max}}}}{c_{y_{r.n}}}.$$

With deceleration of the flight the available overload decreases. In conditions of minimum speed, when  $c_{y_{r.n}} = c_{y_{\text{max}}}$ , the available overload has the least value and is determined by the formula:

$$n_{\text{расч. min}} = \frac{c_{y_{\text{max}}}}{c_{y_{\text{min}}}} = 1.$$

In these conditions the least increase in the angle of attack leads to stalling of the aircraft.

The maximum available overload corresponds to a flight of the aircraft at maximum speed.



$$n_{\text{расч. макс}} = \frac{c_{y_{\text{макс}}}}{c_{y_{\text{мин}}}}.$$

The difference between the available overload and the overload which will be formed in horizontal flight at an assigned speed is called reserve of the overload:

$$\Delta n = n_{\text{расч}} - 1.$$

Overload reserve ( $\Delta n$ ) is the increase in overload necessary for recovery of the aircraft into a critical angle of attack.

The overload reserve in practice can be completely expended by the pilot without a threat of stalling the aircraft. The overload reserve in each condition of horizontal flight of the aircraft can be determined by formula

$$\Delta n = \frac{V^2}{V_{\text{кр}}^2} - 1.$$

To ensure flight safety on the An-2 aircraft the limit of overload reserve is set equal to 0.5.

Factors lowering the value of coefficient  $c_{y_{\text{макс}}}$ , namely, decrease in operating conditions of the engine, icing of the wing, retracting of the flaps, turning off of slats, and also the presence of sideslip, lower the reserve of overload.

#### Limit of Overload with Respect to Strength of the Aircraft

Elements of construction of the aircraft are calculated on strength according to the breaking loads. The magnitude of breaking load for calculations is taken in a definite relationship to the weight of the aircraft.

The ratio of the breaking load to the weight of the aircraft is called the coefficient of breaking overload

$$n_{\text{pas}} = \frac{Y_{\text{pas}}}{\sigma}.$$

In order not to cause on the aircraft permanent deformations and not to destroy the aircraft, loads allowed in flight must be less than breaking loads.

The overload allowed in flight is called operational ( $n$ ); it corresponds to the maximum load ( $Y_0$ ) permissible in flight

$$n = \frac{Y_0}{\sigma}.$$

The number showing in how many times the breaking overload is greater than the maximum permissible operational is called by the safety factor

$$f = \frac{n_{\text{pas}}}{n_{\text{max}}}.$$

The greater the safety factor ( $f$ ), the greater the strength of the aircraft; however, an increase in it is accompanied by an increase in the weight of construction of the aircraft. Therefore, there is set the minimum value of the safety factor at which permanent deformations do not appear with the creation in flight of the maximum permissible operational load. For all aircraft of class "B"  $f = 1.5$ .

In the calculation of strength of the An-2 aircraft there are accepted the maximum permissible operational overloads with respect to the strength of the biplane wing cell: with a flying weight of 5250 kg — 3.74 and with a flying weight of 4740 kg — 4. Proceeding from this coefficient of the breaking overload of the An-2 aircraft will be:

a) for a flying weight of 5250 kg — 5.61 ( $n_{\text{pas}} = n_{\text{max}} \cdot f$ );

b) for gross weight 4740 kg — 6.

## Overloads During Flight in Bumpy Air

In flight under the influence of gusts of air there occurs a change in angles of attack and speed of flight of the aircraft, as a result of which the overload is changed. Under the effect on the aircraft of a horizontal gust of air the overload is increased owing to the increase in local velocity of flow about the wing (Fig. 104a). Under the effect on the aircraft of a vertical gust of air the overload is increased owing to the increase in angle of attack (Fig. 104b). These overloads are well sensed by the pilot. In the case of a positive overload (climb of the aircraft) the body of pilot is pressed to the seat and during negative overload (descent of aircraft) - detached from the seat.

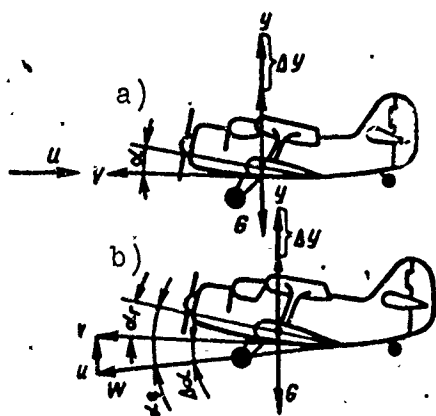


Fig. 104. Appearance of overloads during flight of an aircraft in bumpy air. a) with approach flow of air; b) with ascending flow of air.

The magnitude of the overload during flight in bumpy air can be determined by the formula

$$n = 1 \pm \frac{\rho V W}{G} c_y^a,$$

where the plus sign pertains to case of ascending flow and the minus sign to the case descending flow; the second component (subtrahend) is the increase in overload, which is directly proportional to air density ( $\rho$ ), speed of flight of the aircraft ( $V$ ), speed of the gust of air ( $W$ ) and derivative of the coefficient of lift ( $c_y^a$ ) and inversely proportional to the wing loading ( $\frac{G}{S}$ ).

In the flight of the An-2 aircraft with full flying weight at a speed of 180 km/h at the service ceiling under the influence of a gust of air 10 m/s the overload is increased: with a horizontal gust at 0.25 and with a vertical gust at 0.85.

Thus, the vertical gust causes an increase in overload considerably greater than the horizontal gust of an identical force. With this, as one can see from the formula, an increase in overload will be even greater, the greater the speed in vertical gust and speed of flight of the aircraft (true airspeed) and the less the altitude of flight. Proceeding from this the calculation case of an aircraft is a flight under conditions of bumpy air with presence of vertical gusts.

An increase in flying weight of the aircraft during a flight in bumpy air is accompanied by a decrease in coefficients of breaking and maximum permissible operational overloads, which lowers flight safety, since actual overloads can reach values exceeding the maximum permissible operational. The main danger in the flight of an aircraft in bumpy air with the presence of powerful vertical gusts of air consists in the fact that the aircraft can emerge into conditions of stalling accompanied by losses of altitude, which is especially dangerous in flights at low altitudes.

The center of gravity limits of the aircraft in the flight of the aircraft in bumpy air is especially important. With more tail-heaviness less forces for the change in angle of attack are required and this means for the creation of an overload. Besides this, with the restoration of normal conditions of flight after stalling of the aircraft, the pilot in the presence on the aircraft of centering by the application of insignificant forces to the control column, can create a dangerous overload.

Overloads in flight also have an effect on the organism of the pilot, cause sickly sensations, headache, hemorrhage of the nose, and loss of sight and consciousness. The influence of an overload on the organism of the pilot depends on its magnitude, time of action, recurrence and direction and state of the organism.

### 3. Recommendations for Piloting an Aircraft in Conditions of Turbulence, Thunderstorm Activity and Mountain Terrain

The flight of an aircraft under conditions of bumpy air with the presence of vertical gusts, as was noted earlier, is dangerous not only by the fact that on the aircraft there can be created overloads exceeding the maximum permissible operational overload ( $n_{max}^2$ ), but, mainly, by the fact that in these conditions there can occur a loss in controllability of the aircraft. In this respect there is identical danger in both ascending and descending flows. Descending flows with a vertical velocity over 15 m/s causes thrusts of the aircraft from 100 to 1000 m and more. With such a thrust downwards, the aircraft, not changing its position with respect to the horizon, suddenly loses altitude with rapid accretion of speed, which can exceed the maximum by 30-50%. Such a situation compels the pilot to remove supercharging and pull the control column back, which can lead to the creation of dangerous overloads or the emerging into conditions of pitching down with subsequent transition in steep gliding. In these conditions it is possible to avoid the aircraft from being destroyed only by smooth recovery into horizontal flight.

Smooth recovery of an aircraft from steep glide is permissible without the application of the elevator trim tab at a speed of not more than 220 km/h, since after deflection of the control column for an increase in the angle of attack the speed for a certain time still continues to be increased and can exceed the maximum permissible with respect to conditions of strength of the aircraft. With transition of the aircraft into conditions of a steep glide the speed is rapidly increased; in 11 seconds after pitching down it will be about 200 km/h.

In order to keep the aircraft from entering conditions of pitching down and not to allow the creation of excessive overloads in the presence of powerful vertical flows, which are especially characteristic in zones of thunderstorm activity and mountain terrain the following is expedient:

1) produce horizontal flight at the speed with flying weight of 5250 kg — 180 km/h, and with flying weight of 4740 kg — at a speed of 160 km/h;

2) make turns with a bank up to  $15^{\circ}$  at a speed of 185-190 km/h and only in a horizontal plane;

3) try not to parry all deflections from initial flight conditions under the influence of frequent gusts; hold the control in a position close to the initial balancing, not allowing sharp banks and pitching of the aircraft; react to deflection of the aircraft from initial conditions only in the case when due to the changed pitch angle the speed will start to be increased or decreased as compared to the initial, and with this the control vanes should be deflected as smoothly as possible.

4) in the presence of great bumps do not try to hold the assigned altitude of flight; fly the aircraft by the gyro-horizon and directional gyroscope, since the rate-of-climb indicator will give sudden changes in the readings and will be useless.

Furthermore, for the purpose of providing flight safety one should:

1) if from afar there is seen a thunderstorm overcast, which has clearly expressed borders, make an attempt to fly around it, considering the drift of this overcast by the wind; if, however, such a possibility to bypass the thunderstorm overcast is impossible, return and land at the nearest airfield;

2) during flight over mountain terrain consider the direction and force of the wind, which will form from the windward side of the range powerful ascending air flows and from the leeward side — descending air flows; therefore, in order not to be pressed to summits of the range, fly at an altitude of not less than 1000 m above the relief and start the descent at not less than 20 km from the mountains;

3) during flight in wide canyons fly nearer to the windward side of the slope; in case of descent of the aircraft under the influence of descending air flow do not allow turning downwind, since the aircraft can be thrown to the ground; in the presence of powerful air flows do not enter into narrow canyons under any circumstances;

4) in landing outside an airfield consider that due to turbulence of the air from the leeward side of a mountain the direction of the wind up to an altitude of 100 m is opposite to the motion of air masses in higher layers of the atmosphere.

#### 4. Possible Vibrations on Aircraft and Measures of Combatting Them

Vibrations which can appear on an aircraft in flight are subdivided into three forms: natural, forced and self-excited. Natural vibrations are characteristic for every element of construction of the aircraft; they appear when it receives the initial pulse and have a definite frequency, the magnitude of which depends on the rigidity and mass of the given element of construction.

Forced vibrations occur from the effect of external periodic forces, the source of which can be the unbalanced engine or propeller and the separation of flow from parts of construction of the aircraft. Forced vibrations include oscillations of the frame of the engine, aircraft control rods, instrument panel, caused by operation of the engine and propeller, and also vibrations of the empennage, called buffeting.

Self-excited vibrations occur in the absence of independent forces. Forces maintaining the vibrations appear inside the vibrating system as a result of the presence of the vibrations themselves. Self-excited vibrations include vibrations of the wing and empennage of the flutter type, and they are caused by airflows and deformations. As a result of deformations of the construction the angles of attack change, and the appearing additional aerodynamic forces maintain the vibrations, since the center of rigidity is the point of application of the force of elasticity ( $P_{np}$ ), relative to which additional

aerodynamic forces ( $\Delta Y$ ), applied in the focus of the section of the wing ( $\Phi$ ), and inertial forces ( $P_m$ ), applied in the center of gravity of the section of the wing [CG] ( $UT$ ), torque. The position of the focus, center of rigidity and center of gravity of the section of the wing is shown in Fig. 105.

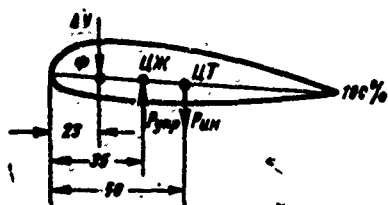


Fig. 105. Location on a chord of the section of the wing of the focus ( $\Phi$ ), center of rigidity [CR] ( $UЖ$ ) and center of gravity (CG).

A vibration of the flutter type is the sharpest and the most dangerous form of vibration; it appears at high speeds of flight exceeding the maximum speed by 30-35%.

For the purpose of preventing the appearance of flutter, on aircraft there is used weight balancing of the controls, which on the An-2 aircraft for ailerons and the rudder - 100% and for the elevator - 105%.

Flutter of the wing and empennage on the An-2 aircraft cannot appear, since its critical speed exceeds the maximum permissible speed with respect to conditions of strength of the aircraft. With the destruction of attachment brackets of balancing weights, which took place during operation of the An-2 aircraft, in flight flexure-aileron and flexure-rudder flutter can appear. With the appearance of rudder flutter one should decrease the speed of flight down to 135-140 km/h and, upon reaching the nearest airfield, land.

Forced oscillations, which can appear on the An-2 aircraft, include vibration buffeting. Buffeting appears during flight of an aircraft at angles of attack close to the critical and appears in the form of impacts about the empennage, which cause shuddering of the aircraft and pulling of the controls.



On certain An-2 aircraft during takeoff, climb and landing there is observed increased vibration of the stabilizer, which is caused by hitting of powerful vortex flow running from the upper wing to the empennage of the aircraft. On such aircraft takeoff and climb must be carried out at nominal or takeoff power of the engine depending upon conditions of the takeoff. In both cases it is necessary to take off with flaps deflected at  $25^{\circ}$ .

The transition to climb after takeoff should be carried out, not changing operating conditions of the engine. At a speed not exceeding 150 km/h, retract the flaps in a two-three procedure, set the cruise setting of operation of the engine and at a speed of 150 km/h continue to climb, not decreasing the climb rate.

In flight, especially under conditions of a bumpy atmosphere, natural wing flutters can lead to a break in the strip braces of the biplane wing cell. In the case of a break in of the carrying strip brace during flight in a bumpy atmosphere, it is necessary to make a forced landing immediately, even away from an airfield. With a break in the carrying strip brace under conditions of a calm atmosphere, one should set the flight conditions which eliminate the vibration of the other strip braces and, avoiding sharp turns, land at the nearest airfield. In case of a break in the rear carrying strip brace during landing one should not deflect the flaps.

A break in the supporting strip brace is not cause for immediate cessation of the flight, since they absorb loads only when standing on the ground and during landing. During landing in this case one should not allow pancaking.

## CHAPTER XV

### BASIC STRUCTURAL-AERODYNAMIC PECULIARITIES AND FLYING CHARACTERISTICS OF THE An-2M AIRCRAFT

#### 1. Structural-Aerodynamic Peculiarities of the An-2M Aircraft

Externally the An-2M aircraft differs from the An-2 aircraft by somewhat modified construction of the fuselage and empennage.

The basic structural changes are subordinate to the problem of providing piloting of the An-2M aircraft by one pilot. For this purpose there are introduced changes in the pilot's cabin and its equipment, the seat of the pilot is raised higher, which considerably improves the view forward and in the right hemisphere, and full remounting of the instrument panel is performed.

Furthermore, the An-2M aircraft has the following structural-aerodynamic peculiarities:

1. The crew's cabin is separated from the cargo and passenger cabin. Separation of the crew's cabin considerably improves conditions of operation of it in the fulfillment of aerial spraying and dusting.

2. When using the aircraft in an agricultural version a more productive special apparatus is installed on it: three-channel duster and rod sprayer, which permit increasing the working width by 30-35%.

Giving a streamlined form to the special equipment made it possible to decrease its drag as compared to the drag of the special equipment of the An-2 aircraft by 25-30%.

The maximum lift-drag ratio of the aircraft when using it in an agricultural version by approximate calculations is 8.7, i.e., somewhat greater than that for the An-2 aircraft.

3. The flying weight of the aircraft is increased up to 5500 kg and the maximum loading in the agricultural version — up to 1500 kg, and in the cargo-passenger version — up to 1400 kg.

4. The freely oriented setting of the tail wheel is replaced by locking in a neutral position, which increases the directional stability during taxiing of the aircraft and increase safety during takeoff and landing.

5. Instead of a manual control there is a foot brake control, which improves operating conditions of the pilot in the controlling of other systems during taxiing.

6. Because of the extension of the wheels of the main landing gears 70 mm forward the anti-nose-over angle is increased, which decreases the possibility of nose-over of the aircraft during taxiing along viscous ground without loading and sharp application of the brakes.

7. The span, area and setting angle of the horizontal empennage are increased as compared to the aircraft An-2 above the 60th series: the span — from 7.2 to 8 m, area — from 12.28 to 15.1 m, setting angle of the stabilizer — from minus 1 to zero. The given structural changes and increase in area of the elevator trim tab permitted increasing the range of the center of gravity of the aircraft because of the increase in tail-heaviness up to 36% of the MAC.

8. The angle of sticking of the ailerons with deflected flaps  $40^\circ$  is increased up to  $20^\circ$ . The angles of deflection of ailerons and the elevator are somewhat modified:

a) the angle of deflection of the ailerons with undeflected flaps is  $30^{\circ}$  upwards and  $15^{\circ}$  downwards;

b) the angle of deflection of ailerons with deflected flaps of  $40^{\circ}$  is  $8^{\circ}$  upwards and  $35^{\circ}$  downwards from the chord of the wing;

c) the elevator is deflected  $40^{\circ}$  upwards and  $22^{\circ}30'$  downwards.

## 2. Flying Characteristics of the An-2M Aircraft

In the process of flight tests the An-2M aircraft with propeller AV-2 at a flying weight of 5500 kg under conditions of ISA (International Standard Atmosphere) indicated the following data:

1. The service ceiling in the transport version is 4100 m and in the agricultural - 3200 m.

2. The practical flying range at an altitude of 1000 m with a cruising speed of 175 km/h with a fuel reserve of 600 kg is 820 km.

3. Maximum speed of flight, km/h

Speeds with respect to altitudes	Aircraft version		
	transport	agricultural	
		with duster	with sprayer
Near the ground.....	226	210	218
At an altitude of 1000 m.....	235	220	223

4. Maximum vertical rate of climb at ground level, m/s

With position of the flaps	Aircraft version		
	transport	agricultural	
		with duster	with sprayer
Undeflected.....	2.6	2.0	2.1
Deflected $5^{\circ}$ .....	2.8	2.2	2.3

5. The fuel consumption per hour in conditions of the greatest duration of flight at a speed of 145 km/h (supercharging 600-650 mm Hg and propeller revolutions of 1700 per minute) is 100-120 kg/h.

6. The fuel consumption per kilometer in conditions of maximum range (speed, 160 km/h, supercharging, 700 mm Hg, revolutions of the propeller, 1700 per minute) is 0.65 kg/km.

7. Cruising speeds in conditions of the greatest cruising power (supercharging, 800 mm Hg, revolutions of the crankshaft, 1700 per minute) are 180-205 km/h.

8. The speed of horizontal flight in the fulfillment of aircraft chemical operations (supercharging, 710 mm Hg, propeller revolutions, 1700 per minute) is 160 km/h.

9. Takeoff characteristics:

a) lift-off speed during takeoff with the use of takeoff power of the engine:

with undeflected flaps — 95-100 km/h;

with deflected flaps of  $30^\circ$  — 70 km/h;

b) takeoff distance:

Aircraft version	Takeoff distance during takeoff, m		
	at rated power		at takeoff power with flaps deflected at $30^\circ$
	without flaps	with flaps deflected at $30^\circ$	
Transport.....	300	210	170
Agricultural with duster...	320	215	180
Agricultural with sprayer..	310	215	175

10. Landing characteristics:

a) landing speed during landing with flaps deflected at  $30^\circ$  -- 85 km/h;

b) landing run on a dry unpaved runway with flaps deflected at  $30^\circ$  and the application of brakes: in the transport version -- 195 m and in agricultural -- 135 m.

11. Satisfactory characteristics of longitudinal stability with respect to overload in the operating speed range. Centering of 33% of the MAC is neutral. During landing on an aircraft with maximum nose-heaviness and deflected flaps of  $30^\circ$ , the reserve of elevator deflection is 42.3%. With departure for a second circle on an aircraft with maximum tail-heaviness and deflected flaps of  $30^\circ$ , the reserve of elevator deflection is 46%.

The character of the behavior, piloting technique and flying operation of the An-2M aircraft basically do not differ from those of the An-2 aircraft.

To ensure the maximum rate of climb climb up to 500 m, it is recommended to accomplish it with flaps deflected at  $5^\circ$  and subsequently without flaps. Deflection of the flaps gives an increase in vertical climb rate of 0.2 m/s. In the determination of the center of gravity by a centering graph, it is considered that in the weight of the construction of the aircraft there is included only the weight of the nondetachable agricultural equipment, and therefore in the presence on the aircraft of tunnel duster the weight of construction of the aircraft must be increased by 60 kg, and to the final result of centering of the aircraft 0.6% MAC must be added.

# A P P E N D I X

TABLE OF UNITS OF MEASUREMENT OF TECHNICAL (MKGSS)  
AND INTERNATIONAL SYSTEMS (INTERNATIONAL  
UNIT SYSTEM)

Designation of quantity	Symbol and formula of definition	Name of the system	
		Technical MKGSS (m-kgf-s)	International Unit System (m-kg-s)
Length	$l$	1 m	1 m
Mass	$m = \frac{f}{j}$	1 and (inertia*) = = 9.80665 kg = kgf·s <sup>2</sup> /m	1 kg
Time	$t$	1 s	1 s
Speed	$v = \frac{l}{t}$	1 m/s	1 m/s
Acceleration	$j = \frac{v_2 - v_1}{t}$	1 m/s <sup>2</sup>	1 m/s <sup>2</sup>
Force	$f = m \cdot j$	1 kgf	1 N (Newton) = 1 kgf·m/s <sup>2</sup>
Work	$A = f \cdot l$	1 kgf·m	1 J (joule) = = 1 kgf·m <sup>2</sup> /s <sup>2</sup>
Power	$N = \frac{A}{t}$	1 kgf·m/s	1 W (watt) = 1 J/s = = 1 kgf·m <sup>2</sup> /s <sup>3</sup>
Pressure	$p = \frac{f}{S}$	1 kgf/m <sup>2</sup>	1 N m <sup>2</sup> = 1 kgf/m·s <sup>2</sup>

\*[Translator's note: this term is not verified].

An analysis of the table shows that of the three basic units in mechanics length (l) and time (t) are common for both the technical, and International systems. Taken as a third basic unit in the technical system is force (f) and in the International system - mass (m).

In the technical system the unit of mass is the inertia - mass of a material particle obtaining acceleration of  $1 \text{ m/s}^2$  under the action of force of 1 kgf. Sometimes this quantity is called t.u.m. (technical unit of mass). The kilogram as a unit of mass differs from the kilogram as a unit of force in that to it is imparted acceleration equal to  $9.81 \text{ m/s}^2$ .

Kilogram-force in contrast to kilogram mass is designated kgf.

All the remaining units forming the system are derivatives.

Conversion from units of measurement of the technical system (MKGSS) into units of measurement of the International System (International Unit System) is carried out with the help of conversion factors:

Units of mass			kg	inertia (t.u.m.)	Units of force	N (Newton)	kgf
1 kg = Inertia (t.u.m.)			1 9.81	0.102 1	1 N = 1 kgf =	1 9.81	0.102 1
Units of work and energy	J	kJ	kgf·m	Units of power	W	kW	kgf m/s
1 J =	1	$10^{-3}$	0.102	1 W =	1	$10^{-3}$	0.102
1 kJ =	$10^3$	1	102	1 kW =	$10^3$	1	102
1 kgfm =	9.81	$9.81 \cdot 10^{-3}$	1	1 kgf·m/s =	9.81	$9.81 \cdot 10^{-3}$	1
Units of pressure	N/m <sup>2</sup>		kgf/m <sup>2</sup>	at		kgf/cm <sup>2</sup>	
1 N/m <sup>2</sup> =	1		0.102	$9.87 \cdot 10^{-6}$		$1.02 \cdot 10^{-5}$	
1 kgf/m <sup>2</sup> =	9.81		1	$9.68 \cdot 10^{-5}$		$10^{-4}$	
1 at =	$1.013 \cdot 10^5$		$1.033 \cdot 10^4$	1		1.033	
1 kgf/cm <sup>2</sup> =	$9.81 \cdot 10^4$		$10^4$	0.968		1	



Conversion from units of measurement of the technical system (MKGSS) into units of measurement of the International System (International Unit System) is produced in the following way:

1. Let us assume that work (A) according to the MKGSS system amounts to 350 kgf·m. To determine the work in units of measurement of systems of the International Unit System it follows:

$$A = 350 \cdot 9,81 = 3433,5 \text{ J}.$$

In order to obtain work expressed in kJ, it is necessary to decrease 3433.5 1000 times ( $10^{-3}$ );  $A = 3.4335 \text{ kJ}$ .

2. Power (N) according to the MKGSS system in technology can be expressed in horsepower (hp), for which the power expressed in kgf·m/s is divided by 75.

To convert, for example, the power of a propeller at 600 hp into units of measurement of the International Unit System the following is necessary:

a) determine how much power will there be in one horsepower in units of measurement of power of the system of the International Unit System:

$$1 \text{ hp} = 75 \cdot 9,81 = 735,75 \text{ W};$$

b) multiplying 600 hp by 735.75 W, we will obtain the equivalent power in units of the system of the International Unit System equal to 441,450 W.

In order to express power in kilowatts, the following is necessary:

$$N = 441\,450 \text{ W} \cdot 10^{-3} = 441,45 \text{ kW}.$$

3. It is known that in technology the technical atmosphere (at) is accepted as the unit of measurement of pressure:

$$1 \text{ at} = 1 \text{ kgf/cm}^2 = 1000 \text{ kgf/m}^2 = 735.6 \text{ mm Hg.}$$

Let us assume that the atmospheric pressure (B) at an airport of departure is 730 mm Hg or 0.96 at according to the MKGSS system.

In order to obtain pressure equivalent to  $B = 730 \text{ mm Hg}$  in units of measurement of the system of the International Unit System, it follows:  $1.013 \cdot 10^5 = 101,300 \text{ N/m}^2$ , and then, multiplying by 0.96, we will determine that

$$B = 730 \text{ mm Hg} = 97248 \text{ N/m}^2.$$

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<p>ABSTRACT</p> <p>(U) The book discusses basic properties and laws of motion of the air, aerodynamic forces of the wing, peculiarities of layout and aerodynamic properties of the aircraft, equilibrium, stability and controllability of the aircraft under different flight conditions, and peculiarities of flight under conditions of icing. The book is intended for students of colleges and schools of civil aviation and can be used by flying personnel of other departments operating the An-2 aircraft. There are 6 tables and 105 figures.</p>					